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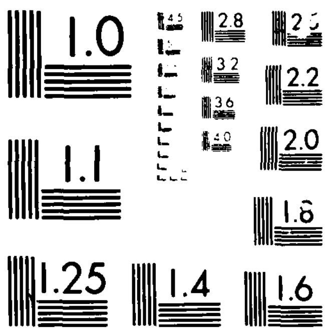

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AD-A168 522

AFGL-TR-85-0310

CONCEPTUAL DESIGN OF A SYNOPTIC INTERPLANETARY MONITOR PLATFORM AT L_1 (SIMPL)

Johns Hopkins University
Applied Physics Laboratory
Johns Hopkins Road
Laurel, Maryland 20707

November 1985

Final Report
16 August 1985 - 30 November 1985

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AIR FORCE GEOPHYSICS LABORATORY
AIR FORCE SYSTEMS COMMAND
UNITED STATES AIR FORCE
HANSCOM AIR FORCE BASE, MASSACHUSETTS 01731

"This technical report has been reviewed and is approved for publication"

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A conceptual design and feasibility study is described for a Synoptic Interplanetary Monitor Platform (SIMPL) to be located at the Earth-Sun interior libration point (L_1), about one million miles from earth. There, SIMPL's magnetometer, solar wind plasma instrument, particle instrument, and radio wave instrument will obtain data to provde the U. S. Air Force Air Weather Service with early warnings of solar-induced disruptions of the terrestrial environment. The data will be processed on-board into militarily useful forecast parameters and transmitted in real-time 24 hours/day to AWS stations worldwide.

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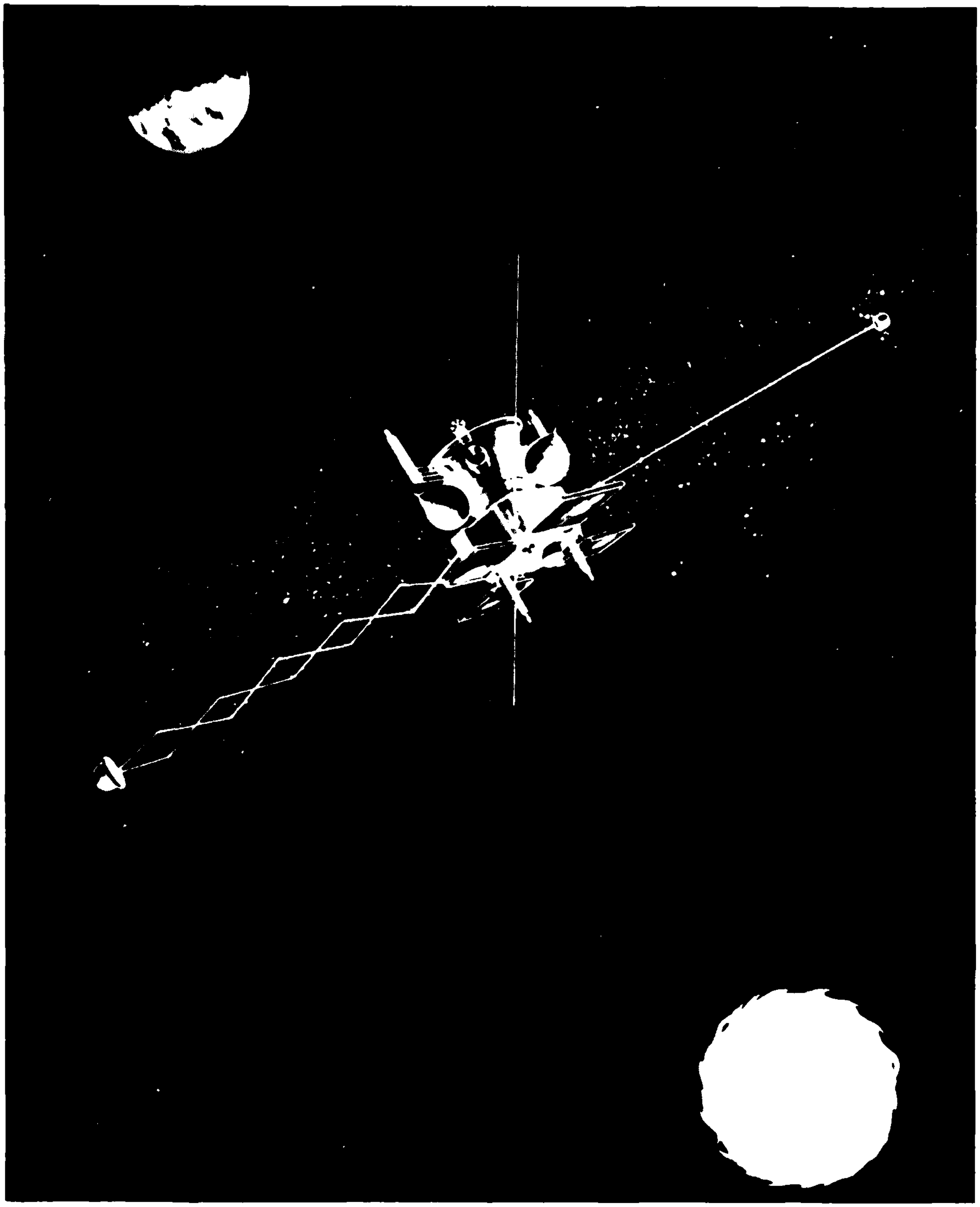
PREFACE

In April 1985 the Air Force Geophysics Laboratory (AFGL) requested expressions of interest in a cost effective libration point monitoring spacecraft. JHU/APL's initial response to that request was contained in report SDO-7599. In August, AFGL funded JHU/APL to perform the mission feasibility study and conceptual design reported here. This report therefore supersedes SDO-7599. A budgetary cost estimate for this program was also prepared and is reported separately.



Artist's concept of the SIMPL spacecraft (thermal blankets not shown)

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1. EXECUTIVE SUMMARY

JHU/APL has performed a feasibility study and conceptual design for an Air Force spacecraft to be located at the L_1 libration point. This spacecraft ("SIMPL") will measure interplanetary parameters and reduce the data onboard to provide advance warning of solar-induced disturbances at the Earth. Real-time data will be returned 24 hours/day to four worldwide receiving stations colocated at existing SOON/RSTN sites. The data will be relayed immediately to Global Weather Central (GWC) for interpretation and dissemination to operational commands.

A halo-type orbit around L_1 has been selected having semi-axis dimensions of 300,000 km by 140,000 km. A spin stabilized spacecraft is recommended, with spin axis approximately parallel to the ecliptic plane. Periodic spin axis precession maneuvers will maintain the +Z axis within 20° of the Sun and the -Z axis within 10° of the Earth. This will permit efficient and economical instrument, power, and RF communications designs.

SIMPL will be Shuttle launched, by direction. We recommend use of the PAM-D upper stage. The PAM-D will yield sufficient extra loft capability and height within its Sunshield to consider launching a subsatellite. Launch constraints and early mission operations have been considered in this study. Shuttle safety requirements have been fully accounted for.

We have determined that a magnetometer, solar wind plasma instrument, particle instrument, and radio wave instrument will meet the Air Force's operational sensor requirements. The instruments will be designed to meet operational, rather than scientific, needs and will generally derive from instruments flown

on previous missions. The magnetometer and the on-board instrument data processor will be fully redundant and the solar wind instrument will be partially redundant.

The spacecraft bus is a modification of our AMPTE/CCE. A conical adapter section has been added to mate to the PAM-D and to provide additional equipment space. Deployables have been limited to the magnetometer boom, its counterbalancing inertia boom, and the two radio wave wire antennas. SIMPL has been designed to fit within the smallest PAM-D Sunshield to minimize Shuttle launch costs.

The solar arrays will be a slightly longer version of those used on AMPTE and will generate 225 W at beginning of life. A Direct Energy Transfer power system operating with a single Ni-Cd battery will provide efficient operation in this 100% Sun orbit. A solar-only backup mode is possible since orbit average load is 70 W less than end-of-life array output and peak demands are low.

The thermal design is also similar to AMPTE's, using multi-layer insulation blankets, thermal coatings, heaters, and possibly one thermal louver. A dedicated radiator will provide additional cooling for the particle instrument.

Attitude detection will be based on redundant star scanners and digital Sun sensors. Attitude will be computed onboard and included in the downlink data product. Active nutation control will stabilize the spacecraft in the early mission phase, prior to boom deployments.

A large, fully redundant hydrazine propulsion system is provided. Made up of Shuttle qualified components, it will

provide a delta-v capability of 790 m/s, versus the 610 m/s needed for a six year mission. Redundant thrusters will provide spin control, spin-axis precession, and orbit adjust.

RF communications will utilize SGLS compatible S-band transponders for both uplink and downlink. Four hemispherical coverge antennas will be used for early mission operations and in emergencies. In the operational (stabilized) mode a high-gain antenna will compensate for the 1,700,000 km slant range and modest (5 W) RF output. Authenticated commanding will take place at 1000 bps. Downlink data at 100 bps will be encrypted and convolutionally encoded. The low bit rate, made possible by the on-board processing, will permit use of small SOON/RSTN receiving antennas, even at this range. UT time will be maintained onboard and included in the downlink. All C&DH and RF hardware will be fully redundant, except for the single high-gain antenna.

Spacecraft magnetic cleanliness will be enhanced by the use of a long (20 ft) magnetometer boom, back-wiring of the arrays, use of a single battery, and other techniques. Special EMC considerations will be included for the radio wave instrument and to meet TEMPEST requirements.

The design of the Ground Support Equipment draws heavily from AMPTE heritage. The required Shuttle Airborne Support Equipment and pre-deployment test philosophy are based on our designs for NASA's ISTP/WIND spacecraft.

We have also performed a conceptual design of the dedicated receiving equipment needed at the SOON/RSTN sites. Transportability, reliability, low cost, and highly automated operation were emphasized. It appears that a 5 m diameter receiving antenna with modest sidelobe control will be

sufficient. A non-autotracking feed and conventional (non-cooled) preamp can be used, reducing cost. The balance of the station hardware will consist of conventional S-band receiving equipment operating under the control of a small station computer. Data will be relayed to GWC and also recorded for backup. No decryption will take place at the SOON/RSTN sites; this will minimize station software development.

SIMPL will be tracked and operated by the AFSCF through its worldwide remote tracking stations. A few station hours/day of SCF support will suffice since the SOON/RSTN sites will be receiving data continuously. Simulations indicate SCF can track and fit a halo orbit. Because SIMPL's slant range greatly exceeds that of any previous SCF mission, some modest SCF station improvements have been identified. Assuming these improvements (or their dB equivalent) are made, adequate RF data link margins exist in all modes at any attitude.

2. SYSTEMS SUMMARY

2.1 MISSION OBJECTIVES

There is a projected military need (Ref. 2-1) for continuous, real-time early warning of ionospheric and geomagnetic disturbances. Forecasts of this type are needed to support DoD satellite operations and manned space activities, especially in polar orbit. Early warning of geomagnetic storms is important to operators of space-track and over-the-horizon radars and long range communications systems. The Strategic Defense Initiative may also require real-time knowledge of the state of the ionosphere and magnetosphere.

Many of the relevant space environmental parameters either cannot be measured from Earth, or provide too little warning time when observed from here. It is especially desirable to make observations from outside the Earth's magnetosphere, to observe the undisturbed solar wind that is the source of so many effects on the Earth. A spacecraft patrolling L_1 can provide at least an hour, and possibly up to two days, warning of solar-induced disruptions. Measurements from L_1 have been unavailable to the USAF Air Weather Service (AWS) since NASA's International Sun-Earth Explorer (ISEE-3) was moved off station in late 1982. Even prior to then, AWS received ISEE-3 data only half the time, on average, and not in real time.

Therefore, the object of this program is to obtain synoptic measurements of interplanetary parameters of importance to the Air Force and return this information in a timely manner to the USAF/AWS. SIMPL will be an operational, not a scientific program, with reliability and cost-effectiveness of prime importance.

2.2 PROGRAM ELEMENTS

The program will contain the following elements:

1. an instrument payload to measure plasma parameters, interplanetary magnetic field, energetic particles, and low frequency radio waves;

2. an on-board instrument data processor to derive militarily useful forecast parameters from the instrument data in real time;

3. a spacecraft bus ("SIMPL") to provide power, attitude control, communications, and other support to maintain the instruments in a halo orbit around the L_1 libration point;

4. a launch vehicle system;

5. a global network of four receiving stations, co-located with existing USAF SOON/RSTN (solar optical and solar radio patrol) stations, to collect SIMPL's forecasting data 24 hours/day and route it to Global Weather Central (GWC, Omaha); and

6. a Mission Control Center (MCC) to monitor spacecraft health, perform orbit tracking and maintenance, and issue commands.

2.3 PROGRAM REQUIREMENTS

The primary requirement is the continuous, real-time transmission of processed forecasting parameters to the four SOON/RSTN stations. For an operational program, the instrument designs must emphasize simplicity and reliability, rather than advancing the sensor state-of-the-art.

The four SOON/RSTN sites assumed for this study are:

	Learmonth, Australia	22°S,	114°E
	San Vito, Italy	40.7°N,	17.7°E
	Ramey AFB, Puerto Rico	18.5°N,	67.2°W
and either	Pelehua, Hawaii	21°N,	158°W
or	Castle AFB, California	37.3°N,	120.4°W.

Receiving antennas installed at these sites may not exceed the largest diameter currently in use, namely 28 ft (about 9 m). The antenna and other dedicated equipment should be sufficiently automated so that minimal support personnel are required, and be portable enough to permit occasional relocation of the station.

The SIMPL spacecraft must orbit in the vicinity of the L_1 libration point and be designed for a Shuttle launch. Mission lifetime is four years minimum, six years goal. A complete ground spare spacecraft is required.

Both raw and processed sensor data must be encrypted. However, the useful life of the data will not exceed a few days. Data are not required to be decrypted or displayed at the SOON/RSTN receiving sites. The spacecraft uplink is required to be authenticated (spoof proof).

Command, control, and tracking of the spacecraft may <u>not</u> depend on NASA's Deep Space Network (DSN) for primary support. This effectively dictates the use of the Air Force Satellite Control Facility (SCF), which was designed primarily to support near-Earth spacecraft.

We assume that SIMPL need not be hardened against nuclear or physical attack due to the inaccessibility of its orbit. SIMPL will be fully resistant to the natural radiation environment, as outlined in Appendix B.

Finally, but most important, is the need to accomplish this mission in a cost-effective manner. We have sought to minimize not only the spacecraft cost, but the entire life cycle cost of spacecraft, upper stage, ground stations, and support.

3. INTERPLANETARY MONITOR PAYLOAD

The basic aim of the SIMPL spacecraft is to provide a continuous monitor of selected parameters in the space environment to give advance warning of solar-terrestrial disturbances. These disturbances may affect the performance of spacecraft, communications, manned space activities, and over-the-horizon radar. SIMPL will provide data to help predict the occurrence of geomagnetic storms, sudden magnetic impulses, energetic proton events, and enhanced heavy ion fluxes. These phenomena not only affect the performance of communications and detection systems, they may make manned space activities unsafe. Heavy ion fluxes may affect the stability of operating space hardware with single event upsets and other radiation related effects.

The existing body of scientific results from such NASA spacecraft as ISEE-3, IMP-7, and IMP-8 has shown that by monitoring several critical parameters in the space environment, the forecasting of solar-terrestrial disturbances can be greatly improved. The vital data include the interplanetary magnetic field vector, the solar wind plasma parameters, and the energetic particle fluxes.

The build-up process for geomagnetic storms is closely linked to the southward component of the interplanetary magnetic field (B_Z). While the details of the substorm process are still a very active area for scientific research, B_Z is clearly an important factor and advance knowledge of B_Z will significantly improve forecasting. The transfer of energy from the solar wind to the terrestrial magnetosphere involves the magnetic field, solar wind velocity, and solar wind density. This is another area

where active scientific work continues, but the present state of knowledge is already sufficient to improve forecasting accuracy if the data were available.

A monitoring spacecraft could detect the onset of high speed solar wind streams and interplanetary shock waves with a warning time roughly proportional to its distance from the Earth along the Earth-Sun line. For SIMPL at the L_1 libration point, this warning time is about one hour.

Although no advance warning of energetic particles from in-situ measurements is possible because of their velocity, a spacecraft at L_1 is in an ideal position to monitor the energetic particle flux reaching the terrestrial magnetosphere. Recent scientific results, however, show that low frequency radio observations of the Sun can provide good indicators of energetic solar particle injection from the corona and the approach of interplanetary shock waves. Thus a low frequency radio instrument has the potential for providing advance warning on the order of an hour for energetic particles and a few days for interplanetary shocks.

3.1 PREDICTION DATA REQUIREMENTS

The highest priority data for SIMPL are samples of the interplanetary magnetic field vector. The magnetometer should provide the vector components and possibly an indication of magnetic field turbulence.

Solar wind plasma measurements are required with emphasis on the quality of solar wind speed and density. The solar wind temperature is of lesser importance. Additional solar wind parameters may be available at essentially no additional cost.

Parameters, such as the helium velocity or heavy ion composition, are not considered primary data types at this time but may prove useful in the future.

Energetic particle measurements should include fluxes over a wide energy range. Protons, heavy ions, and electrons should be measured. Additional parameters, such as energy spectra and anisotropy magnitudes, may be available without additional effort.

Thus, a magnetometer, solar wind plasma instrument, and an energetic particle instrument have been included as the primary instrument complement for SIMPL. The solar wind instrument may be either a Faraday cup or an electrostatic analyzer.

Low frequency radio measurements have the potential for providing the greatest warning time for energetic particles and traveling interplanetary phenomena; however, the use of such data is the least well understood of the basic measurement types. Therefore, a low frequency radio instrument has been included as a secondary data source on SIMPL.

3.1.1 Time Resolution of Data

For SIMPL to be useful as an advanced warning and monitoring spacecraft, the data must reach the Air Force Global Weather Central (AFGWC) in near real time. Those data which have a predictive role must be available in a fraction of their advance warning time with an appropriate time resolution. Since the warning time is on the order of one hour, a time resolution of five minutes is reasonable if the data are rapidly transferred from the ground receiving station to AFGWC. Five minute resolution is also reasonable from the scientific standpoint for

most monitoring parameters. Magnetic field vectors and some particle fluxes, however, may be useful at a higher time resolution. Therefore, the magnetic field vectors are planned to be read out at four vectors per minute, and some fluxes will be read out at one minute intervals, while all the other primary data have a five minute time resolution. Some of the secondary data from each instrument may have a time resolution that is much more coarse.

3.1.2 On-Board Processing

The SIMPL instrument data must be put into "scientific units" before they can be used for prediction. For the magnetic field, this means calibrated vector components in nanoteslas along a useful coordinate system, such as the Geocentric Solar Ecliptic (GSE) system. Similarly the plasma distribution must be reduced to velocity, density, and temperature, and the particle data must be calibrated in flux.

This data reduction could be done either on the ground or in the spacecraft. However, there are severe data rate limitations for a spacecraft at L_1 that must transmit to small receiving stations on the ground. These limitations would severely limit the voluminous quantity of raw data that must be processed to produce reliable five-minute averaged prediction parameters.

Ground data processing also represents a significant operational cost for the duration of a long-lived mission. It is often less expensive to build the processing into the spacecraft.

Therefore, SIMPL has baselined that all routine data processing will be done onboard. A central instrument data processor will take the instrument outputs and provide the additional processing required for each data type. This would nominally include determining the moments of the plasma distribution, de-spinning and transforming the magnetic vectors into GSE coordinates, averaging particle fluxes, etc.

3.1.3 Instrument Orbit/Attitude Requirements

There is only one instrument requirement on the spacecraft orbit dimensions. The spacecraft angular distance from the Earth-Sun line should be kept as small as possible when viewed from Earth. This will provide the best correlation between the environmental data from SIMPL and the actual plasma and magnetic field vectors that arrive at the terrestrial magnetosphere about one hour later. This desire to be close to the Earth-Sun line must be balanced against the requirements of orbital mechanics and the need for a sufficient Sun-Earth-spacecraft angle so that reception at the ground stations is not impaired by radio noise from solar activity.

The spacecraft attitude requirements are determined by a number of instrumental constraints. Most instruments operate best on a spinning spacecraft. A spinning magnetometer can easily be corrected for offsets in two axes. Electrostatic analyzer types of solar wind instruments operate best when their fields of view sweep over the Sunward direction. Energetic particle instruments can cover a wide range of particle pitch angles from a spinning spacecraft without the need for any moving parts. And radio instruments can be used for direction finding of interplanetary sources, if the antennas spin in the ecliptic plane.

The angle between the spacecraft spin axis and the Sun, along with its maximum deviation from the nominal value, are also important. Faraday cup style solar wind instruments have an acceptance cone that is limited to approximately 20° for a single cup. Electrostatic analyzer design also becomes much more difficult if the Sun-angle range is large. The energetic particle instrument operates best over a limited range in Sun-angle, so that pitch angle coverage can be optimized.

Finally, it must be realized that space-proven instruments exist only for a limited range of spin-axis options. Four possible orientations were considered for SIMPL, and a somewhat subjective rating system was used to evaluate the merits of each orientation for each instrument. The results are shown in Table 3-1. Each instrument has been assigned an importance value ranging from 60% to 100%, and a value has been used to summarize the overall ease of providing the required SIMPL data for each orientation. Weighted totals for each of the possible orientations are listed along the bottom row of the table. A spin axis perpendicular to the ecliptic is the first choice, and a three-axis stabilized spacecraft is clearly the least desirable. The second best choice, with the spin axis pointing toward the Earth, is nearly as good but it does not support radio wave direction finding and there is some reduction in pitch angle coverage. This orientation, however, is by far the best orientation for spacecraft performance and low cost, as shown in Section 4.3. Thus, a spin axis that points approximately toward the Earth, with a limit on the spacecraft-Sun angle, has been chosen as the baseline for SIMPL.

Table 3-1

SIMPL spin-axis options

	3-Axis Stablized (Voyager, SOHO)	Spin ⊥ Ecliptic (ISEE-3, WIND)	Spin ~ Toward Sun (Tipped WIND, WAMPTE)	Spin ~ Toward Earth
Solar Wind Instrument (Importance = 100%)	Requires complex instrument 75%	Standard instrument 100%	Complex instrument 75%	Existing instrument 90%
Magnetometer (Importance = 100%)	Difficult to remove bias and offset 75%	Standard instrument 100%	Standard instrument 100%	Standard instrument 100%
Energetic Particles (Importance = 80%)	Poor pitch angle coverage 50%	~ Full coverage 100%	Modest pitch angle coverage 80%	Modest pitch angle coverage 80%
Radio Wave (Importance = 60%)	Antenna problems 60%	Best for direction finding 100%	Poor directionality 75%	Poor directionality 75%
Instruments Weighted Value	2.26	3.40	2.84	3.00

3.1.4 Instrument Data Allocations

Real-time data collection with small receiving antennas limits the total SIMPL data stream to approximately 100 bits per second. This limited bandwidth must support all of the processed data, the supporting raw data from the instruments, spacecraft attitude sensor data, and all housekeeping functions. The baseline data allocations are listed in Table 3-2. The processed data along with the housekeeping status flags and alarms total only 20 bps. However, even this very low data rate will accumulate ~ 100 million bytes of data per year. The total data stream of ~ 1 megabyte per day will be recorded for later analysis.

3.1.5 Post Processing of SIMPL Data

The recorded SIMPL data stream will be distributed to the instrument teams and the spacecraft operations team. The operations team will use the raw attitude data to confirm the quality of the on-board attitude solution. They will also check the housekeeping data for spacecraft health and safety. A memory trickle readout has been planned to confirm the proper operation of the various spacecraft data processors as well.

The instrument teams will use the recorded data stream to check the operation of the instruments and the data reduction both within the instruments and in the central instrument data processor. Calibration of the processed data stream will be checked. The raw data may also be used for post-event analyses to improve the predictive quality of the instrument data. If improved data processing algorithms are developed as a consequence of these analyses, software in the central instrument processor may be modified.

Table 3-2

SIMPL instrument data rates

	Data for Real-Time Use	Data Buffered for Post Processing
Magnetometer		
Processed Data	5 bps	5 bps
Raw Data		8
Solar Wind		
Processed Data	1	1
Raw Data		16
Energetic Particles		
Processed Data	5	5
Raw Data		13
Radio Receiver		
Processed Data	4	4
Raw Data		8
Attitude Data		10
Housekeeping		10
Memory Trickle		10
Header, Overhead		10
Total Data Rate for Real-Time Use	15 bps	Total Data Buffered for Post Processing 100 bps

1.08 Megabytes/Day

The SIMPL data stream can be stored until it fills the tape, which can then be mailed to the investigator teams. Typical low-cost tape cartridges can hold from one to three months of data. However, a post-processing delay of more than a month is a poor choice for validating data quality.

3.2 INSTRUMENTS

Studies of the instrument requirements have been carried out with the assistance of scientific research groups that have extensive experience with each of the four instrument types. Specific contacts for each instrument are: Magnetometer, Dr. L. J. Zanetti of The Johns Hopkins University Applied Physics Laboratory; Solar Wind Plasma, Drs. S. J. Bame and R. D. Zwickl of Los Alamos National Laboratories and Dr. A. Lazarus of the Massachusetts Institute of Technology; Energetic Particle Instrument, Dr. R. E. Gold of The Johns Hopkins University Applied Physics Laboratory; Radio Receiver, Drs. R. Stone and J. Fainberg of the Goddard Space Flight Center.

3.2.1 Magnetometer

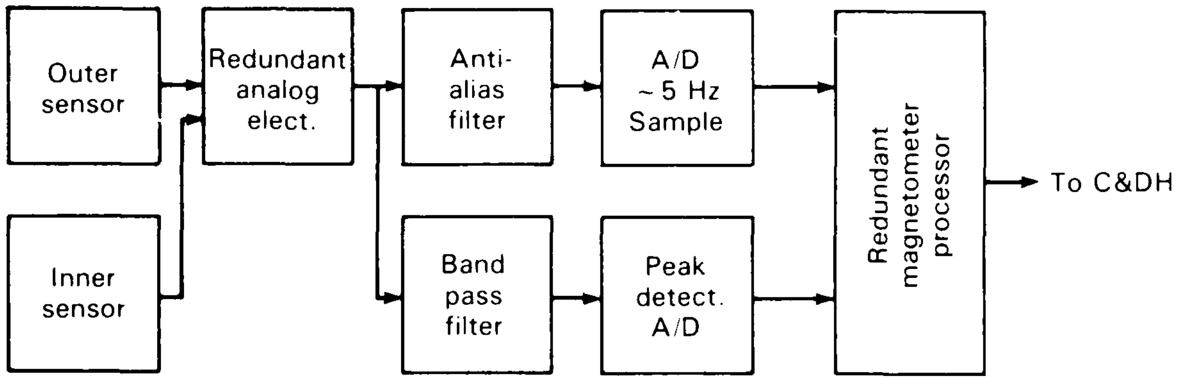
The SIMPL magnetometer will measure nominal interplanetary field strengths of ~ 5 nanoteslas. Therefore a background field, from the spacecraft, should be less than 0.1 nanotesla. A 20 ft magnetometer boom has been incorporated to ensure that spacecraft stray fields are at an acceptable level. Stray fields generated at the center of the spacecraft will be reduced by a factor of 330 at the magnetometer location.

Since the magnetometer is of the highest importance, the SIMPL baseline carries a fully redundant magnetometer system. One sensor is at the end of the boom and the other is fractionally inboard. The two systems may be switched on command. This dual magnetometer system has the additional advantage of being able to use the two sensors at different distances from the spacecraft to improve the estimate of stray spacecraft fields.

A block diagram of the magnetometer is shown in Fig. 3-1. The primary magnetometer output is a despun magnetic field vector every 15 seconds to the central instrument data processor. The magnetometer will also provide a measure of the field noise in two spectral bands. The raw data will consist of snapshots, each covering a 15 second sample period, that are trickled out of the magnetometer memory once every 10 minutes. This memory could also be triggered by strong events, such as interplanetary shocks passing the spacecraft. The magnetometer mass is 4.6 kg and the average power consumption is 2.2 watts.

3.2.2 Solar Wind Plasma Instrument

SIMPL has been configured so that it is compatible with both electrostatic analyzer and Faraday cup instruments. The solar wind instrument and the magnetometer are the most vital to the success of SIMPL. Therefore, redundancy has been included in these two baseline instruments. However, because of the mass and volume of the solar wind instrument, only partial redundancy has been included. The baseline solar wind instrument has redundant high voltage supplies and processors but only one sensor head. This instrument could be made fully redundant for an additional penalty in mass and volume which may impact other instruments or subsystems.



Weight ~ 4.6 kg
Power 2 watts

Data outputs:

	Data rate Direct	Total
Despun vectors ~ 4/min	4.4	4.4
Trickle of raw A/D output and events		7.6
BNOISE in ~ 5-50 Hz @ 1/min	.25	.25
BNOISE in .1-5 @ Hz @ 1/min	.25	.25
Housekeeping (may be partially analog)	.1	0.5
Total	5.0	13.0

Fig. 3-1. SIMPL magnetometer block diagram.

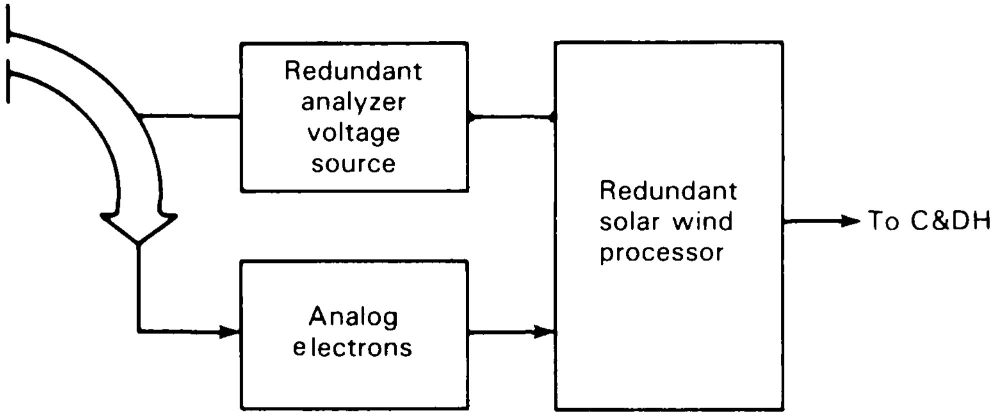
Figure 3-2 is a block diagram of the solar wind instrument. The instrument determines the energy per charge spectrum of the solar wind and organizes the data within its own processor. A subset of these data is passed to the central instrument processor which fits it to a convected Maxwellian distribution and determines the solar wind velocity, density, and temperature of five minute averages. This instrument may also occasionally scan the helium and heavy ion portion of the spectrum and trickle these data down with the raw data.

The SIMPL orbit and attitude can be accommodated with existing or planned Faraday cup and electrostatic analyzer instruments. The MIT Faraday cup instrument planned for the NASA WIND mission would also work on SIMPL. The existing Los Alamos electrostatic analyzer that is on the NASA Ulysses mission (formerly called the International Solar Polar Mission) will work over a range of Sun angles from 0° to 60°, which surpasses the SIMPL requirements of 5° to 20°. Because it is an existing instrument, the Los Alamos design has been used as the baseline for SIMPL.

The Los Alamos instrument could be simplified with fewer channeltrons and smaller size if it were optimized for SIMPL. However, it may be less expensive to use the existing design and possibly even the Ulysses spare instrument. The SIMPL baseline allots 5 kg and 4 watts for the solar wind instrument.

3.2.3 Energetic Particle Detector

The energetic particle detector for SIMPL (Fig. 3-3) must cover a wide range of energies for ions and electrons. An instrument similar to the JHU/APL Charged Particle Measurements Experiment on the IMP-8 spacecraft will provide the required

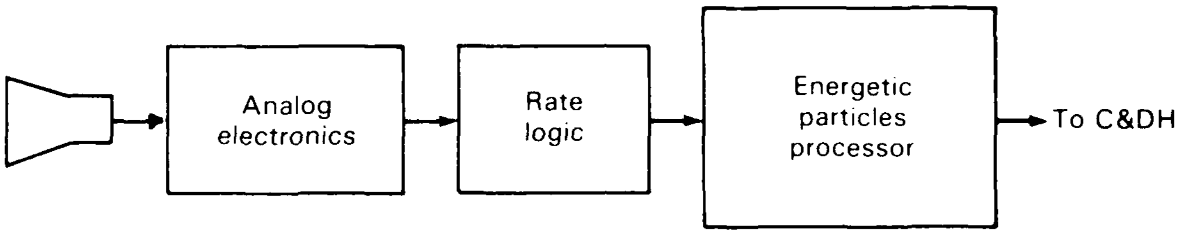


Weight 5 kg
Power 4 watts

Data outputs (After processing by S/C data processor):

	Data rate (bps) Direct	Total
Solar wind velocity (5 min)	.25	.25
Solar wind density (5 min)	.25	.25
Solar wind temperature (5 min)	.25	.25
Raw data in H/He mode		13.0
Raw data in heavy ion mode		3.0
Housekeeping	.25	.25
Total	1.0	17.0

Fig. 3-2. SIMPL solar wind plasma instrument block diagram.



Weight 5 kg
Power 5 watts

Data outputs:

	Data rate (bps) Direct	Total
Ion fluxes (5 min)	2.5	2.5
Electron fluxes (5 min)	.5	.5
Energy spectrum (1 hr)	1.0	1.0
Anisotropy magnitude (5 min)	.5	.5
Raw data		12.5
Housekeeping	.5	1.0
Total	5.0	18.0

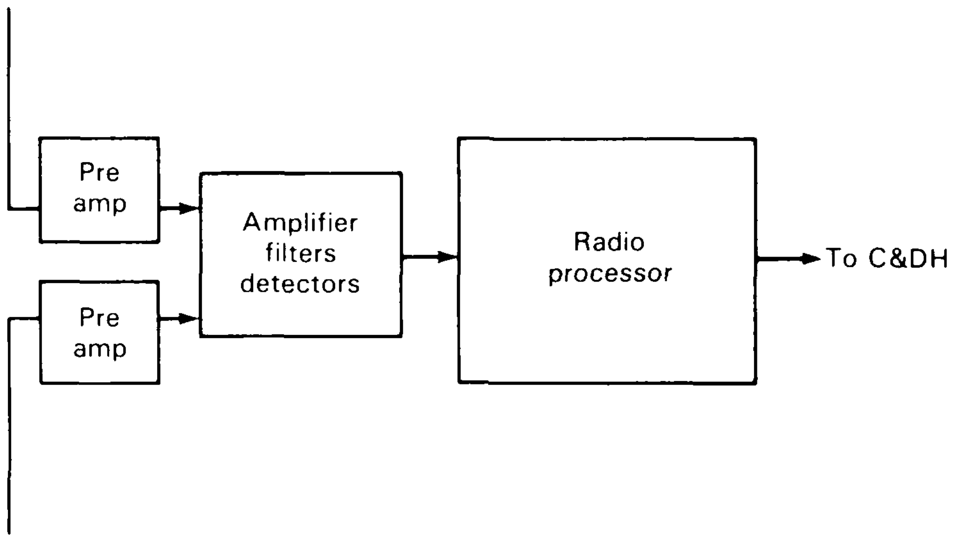
Fig. 3-3. SIMPL energetic particle detector block diagram.

data. This is a three element silicon solid state detector stack surrounded by an anticoincidence cup. The energy range of the instrument covers ~ 0.1 MeV to ≥100 MeV for ions. The anticoincidence cup also provides a detector with a large geometry factor for cosmic rays and intense solar flares with a threshold energy of ~70 MeV. There are also energetic heavy ion channels ($Z \geq 3$, $Z \geq 10$) to detect particles that are likely to cause single event upsets in space electronics. Electrons will be measured in the energy range from 0.2 to 2.5 MeV. The geometry factor of this instrument is sufficiently large to make statistically meaningful measurements of very small events in five minutes while the counting electronics can handle very large events without saturation.

The energetic particle detector may also provide a detailed energy spectrum and anisotropy measurement at much lower time resolution. The particle instrument is allocated 5 kg and 5 watts.

3.2.4 Radio Receiver

In order to provide the greatest warning time for interplanetary disturbances and solar particle events, the radio receiver should cover a wide frequency range. The baseline SIMPL instrument will work from ~ 10 kHz to 1 MHz. The existing Ulysses design is basically compatible with the SIMPL requirements, but it is far more comprehensive than required for SIMPL. Several subsystems of the Ulysses instrument (including the plasma sounder, Fourier transform electronics, etc.) would be eliminated, and the number of antennas would be reduced to two. Figure 3-4 is a block diagram of the radio receiver. The mass of the instrument is 8.4 kg, including the antennas, and it uses 5.6 watts.



Weight	3.4 kg	Electronics
	5.0 kg	2 wire antennas
	8.4 kg	

Power: 5.6 watts

Data outputs:

	Data rate Direct	Total
Processed frequency band flux and events	4.	4.
Raw data		8.
Total	4.0	12.0

Fig. 3-4. SIMPL radio receiver block diagram.

Data outputs from the radio receiver would consist of frequency-time spectrograms which would be reduced to a much simplified output stream by the central instrument data processor.

4. MISSION DESIGN

The type of orbit, orbit dimensions, method of ascent, and spacecraft spin axis orientation are all interrelated. We have examined a broad range of possibilities to arrive at the best approach for SIMPL.

4.1 THE L_1 LIBRATION POINT

To study the flow of energy from the Sun through the interplanetary medium and into the Earth's magnetosphere, it is desirable to position instruments along the Earth-Sun line. The Earth-Sun interior libration point, L_1, is located on that line 1.5×10^6 km (about a million miles) from Earth. At L_1 the forces on a spacecraft balance to zero, and it can be maintained orbiting that point. Instruments located at L_1 can detect and warn of solar-induced phenomena several hours before the aerospace environment of the Earth is disturbed.

The launch (in August 1978) of ISEE-3, followed by a very successful mission, proved the practicality of maintaining a spacecraft in a halo orbit around a libration point. After nearly four years at L_1, ISEE-3 departed to begin its historic comet intercept mission. Since 1982 the important L_1 region has remained unmonitored.

4.2 L_1 ORBIT OPTIONS

The L_1-centered coordinate system used in discussing libration point orbits is shown in Fig. 4-1. The x-axis is along the Earth-Sun line, positive toward Earth. The z-axis is normal to the ecliptic plane, positive to the north. The y-axis lies in the ecliptic plane, completing the right-hand system.

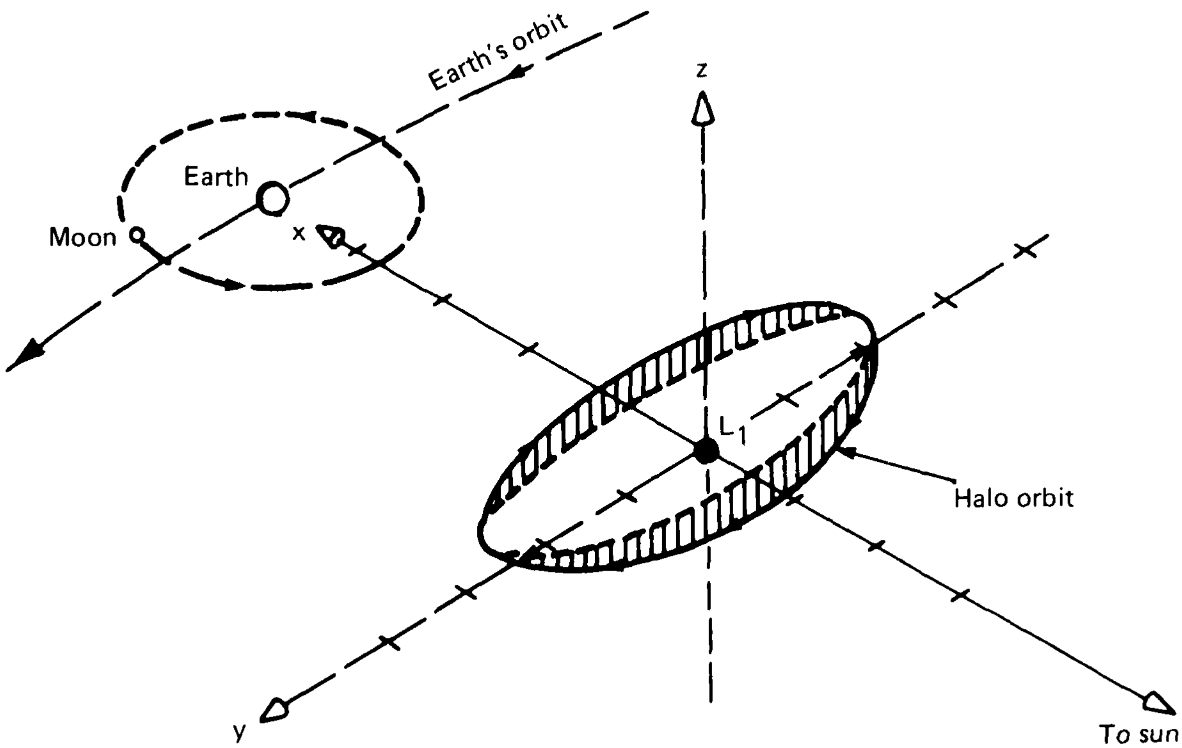


Fig. 4-1. L_1 coordinate system and halo orbit (after Farquhar, Ref. 4-1).

A spacecraft orbiting L_1 experiences harmonic motion along x, y, and z. The x and y motions are coupled; the z motion is essentially independent. When viewed from Earth, the trajectory has the appearance of a Lissajous pattern. If the z-axis fundamental frequency is set equal to that of the x-y motion, a true "halo" (single closed loop) orbit results.

L_1 orbits can be classified by size according to their "semi-major axis," A_y. A_y defines the greatest distance from the Earth-Sun line, in the ecliptic plane. For a true halo orbit to exist, A_y must exceed 654,000 km. If a halo type orbit is desired with an A_y smaller than this, it can be approximated by a "broken Lissajous" in which orbit corrections are made every month or so (see Ref. 4-1). L_1 orbits considered by mission planners to date have ranged from very small (A_y=120,000 km) up to "large" (A_y=800,000 km). ISEE-3 was placed in a true halo orbit at A_y=667,000 km. The period for any of these orbits, regardless of size, is about 178 days.

Any type of orbit around a colinear libration point is inherently unstable and needs periodic correction thrusts to maintain it. The smaller A_y orbits require more delta-v (velocity change capability) for both orbit maintenance and initial orbit insertion. The range of insertion delta-v is from 50 to 300 m/s depending on A_y, launch date, permissible travel and insertion time, and whether or not lunar swingby gravity assist is employed during ascent. Below A_y=654,000 km, the need for period control causes maintenance delta-v to rise slightly with decreasing A_y. An additional maintenance delta-v term is directly proportional to the out-of-plane amplitude, A_z, at about 17 m/s/year per 100,000 km.

4.3 SPIN-AXIS ORIENTATIONS

Orbit dimensions and spin-axis orientation must be selected together. We examined four possible spin-axis orientations for SIMPL.

<u>Three-Axis Stabilized</u> - in which a non-spinning spacecraft would maintain one face "staring" at the Sun;

<u>Spin Normal to Ecliptic</u> - in which the spin axis would be maintained normal to the ecliptic plane, like ISEE-3;

<u>Spin Toward the Sun</u> - in which the spin axis would lie approximately parallel to the ecliptic plane and be maintained within a small angle to the Sun; and

<u>Spin Toward the Earth</u> - in which the spin axis would lie approximately parallel to the ecliptic plane and be maintained within a small angle to the Earth.

These four options were examined on the basis of system complexity and sensor effectiveness. In Section 3 we showed how each instrument fared under each spin option and then weighted those scores by each instrument's importance to SIMPL. We did likewise for those spacecraft systems whose performance, complexity, or cost are spin-dependent. The results are summarized in Table 4-1. While admittedly somewhat subjective, this approach allowed us to draw some important conclusions.

We found that the three-axis stabilized option could be ruled out immediately. It was least desirable for the instruments (most of which benefit from spin) and led to the most complex and

Table 4-1

Compatibility of major spacecraft subsystems
with the four spin-axis options

	3-Axis Stabilized	Spin ⊥ Ecliptic	Spin ~ Toward Sun	Spin ~ Toward Earth
Communications (Importance = 100%)	Articulated antenna, Highest gain 40%	Expensive, Low-gain pancake beam 40%	Small dish, Modest gain 70%	Small dish, High gain 90%
Solar Panels (Importance = 100%)	Fewest cells 100%	π:1, More cells 30%	Good efficiency 90%	Good efficiency 90%
Attitude Detection (Importance = 100%)	Star cameras 30%	85%	Requires star scanner 60%	Requires star scanner 70%
Attitude Control (Importance = 100%)	Momentum wheels 20%	100%	90%	90%
S/C Weighted Value	1.9	2.55	3.10	3.40
Instruments Weighted Value (from Section 3)	2.26	3.40	2.84	3.00
Composite Score	4.16	5.95	5.94	6.40

expensive spacecraft. In the area of attitude detection and control the absence of spin implied a need for a star tracker or camera rather than the simpler star scanner. It also required momentum wheels (and momentum dumping), ruled out use of simple bladderless propulsion tanks, and made it more difficult to stiffen wire booms.

The spin-normal-to-ecliptic option had the highest instrument score by a small margin. In addition, it permits selection of a high A_y orbit with low on-board propulsion requirements, provided a toroidal pattern ("pancake beam") antenna is used. However, pancake-beam antennas provide only moderate gain, requiring larger dishes at the SOON/RSTN stations. A pancake-beam antenna, such as used for ISEE-3, is expensive, provides too little gain, and its height is difficult to accommodate in the Shuttle. In addition, the electronic beam switching needed to provide two or more tilted pancake beams introduces RF losses and increased risk of single-point failure. A mechanically or electronically despun antenna could provide sufficient gain, but at even greater cost, complexity, and risk.

A second major objection to this spin option is that less than one-third of the maximum solar array output is available at any one time. Solar arrays are a major cost and weight item, so it is preferable to fully illuminate a smaller array all the time. This spin option would rule out use of our small, low-cost AMPTE bus since a four-sided spacecraft could not reasonably be adapted to this spin orientation. (AMPTE is described briefly in Appendix A.) It would drive the design toward the larger WIND spacecraft we designed for NASA's ISTP program, where spin normal to the ecliptic was dictated because of a different set of science requirements. WIND would be more expensive not only for the

reasons cited above, but also because it represents a new design. We therefore concluded that the spacecraft disadvantages strongly outweighed the small instrument advantage for the spin-normal-to-ecliptic option.

The last two options considered differ only slightly. In the Sun-pointing option, the spin axis, fixed in inertial space, would be initially set to some maximum permissible negative Sun angle, such as -20°. As L_1 moved around the Sun at roughly 1°/day, the Sun angle would drift through zero and then increase toward positive values. When it reached, say, +20°, the spacecraft spin axis would be torqued so as to reset the Sun angle to -20°, and a new drift arc would begin.

In the Earth-pointing option, the angle between the spacecraft -z axis and the Earth would be the one maintained within bounds. Figure 4-2 illustrates the Earth-centered option. The delta-v needed to rotate the spin axis one complete turn per year is negligible (<1 m/s/year) for the angular momentum we anticipate for SIMPL.

We found that from an instrument point-of-view, these two options were nearly equivalent, provided maximum Sun angle could be maintained below about 20° for the solar wind instrument. Either option permits basing the SIMPL spacecraft on our AMPTE bus, thereby raising the possibility of a truly low-cost mission. AMPTE was designed to keep one face approximately toward the Sun; its power and thermal systems can easily tolerate ±20° Sun angle.

The major advantage of the Earth-pointing option is the opportunity to achieve high gain from a simple, reliable, and low-cost pencil-beam transmitting antenna. This will permit use of

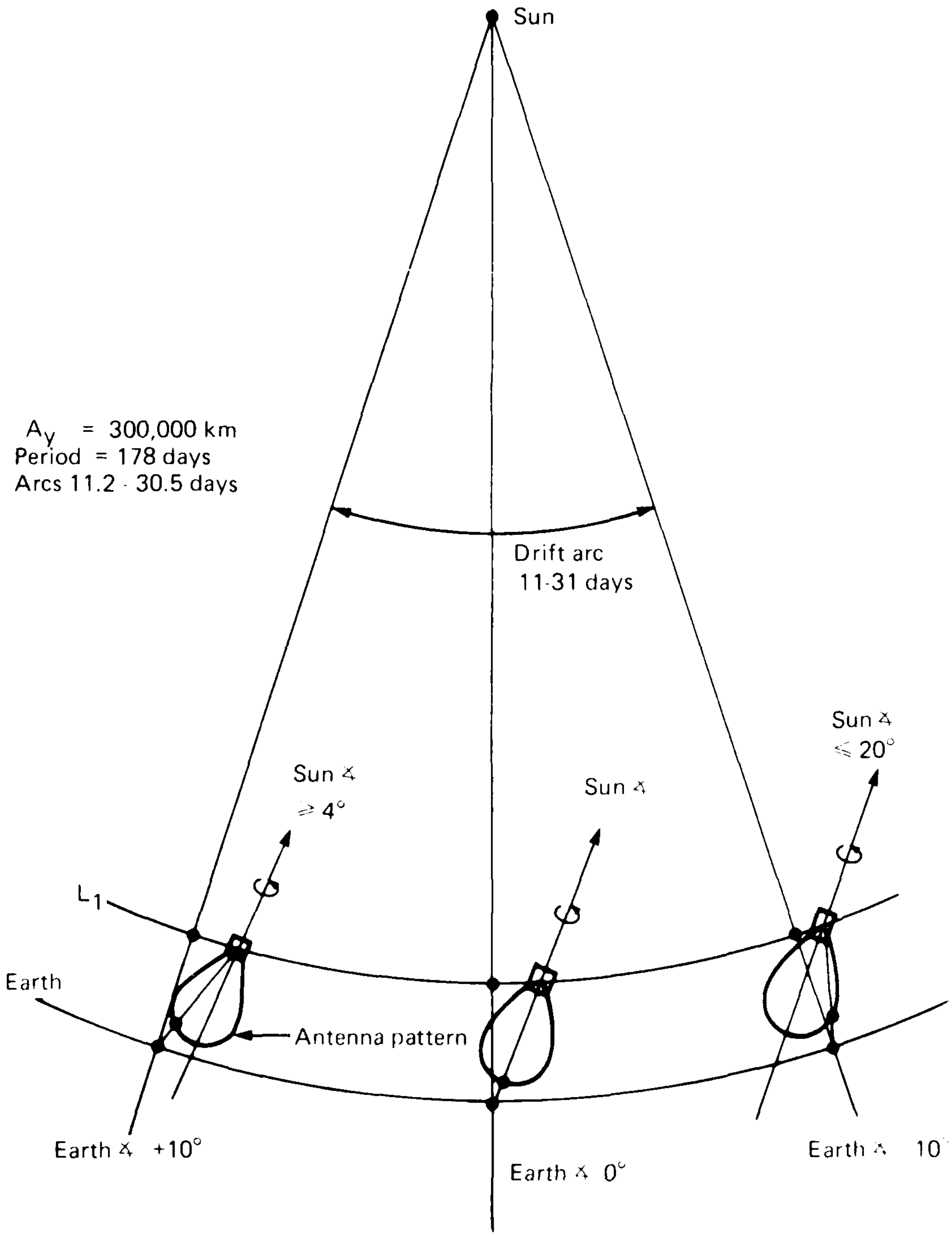


Fig. 4-2. A single drift arc of the Earth-centered orbit option.

the smallest antennas at the SOON/RSTN stations, thereby reducing costs in both the space and ground elements simultaneously. For example, with Earth angle constrained to <10°, a minimum gain of 14 dB can be achieved, compared with only 6.6 dB from the more complex and expensive ISEE-3 type antenna.

Considering all these factors, we therefore selected a spin scenario in which SIMPL's -z axis is maintained within a small angle of the Earth.

4.4 ORBIT DIMENSIONS

Having selected a spin orientation, it remains to select orbit dimensions. The constraints on the orbit are as follows.

Sun Angle (angle between +z spin axis and the Sun) must be kept <20° for the benefit of the solar wind instrument and the power and thermal systems. In addition, it is desirable that Sun angle not go below about 4-5° to permit use of simplified Sun sensors.

Earth Angle (angle between -z spin axis and the Earth) must be kept as small as possible to maximize transmit antenna gain. Above about 20-25°, the pencil-beam antenna begins to lose its advantage over a tilting, pancake beam antenna. Below about 6°, the antenna diameter at S-band becomes too large for AMPTE.

Solar Elongation Angle (angle between Sun and spacecraft, as viewed from Earth) must exceed 5°. Although the RF Sun at S-band subtends only 0.75° when viewed from Earth, solar noise inputs to the sidelobes of ground antennas can degrade downlink signal-to-noise ratio. Small

antennas are somewhat more sensitive to this effect than large antennas. We examined the situation for a worst case Sun (1000 SFU) and concluded that by properly controlling sidelobes, we could operate small SOON/RSTN dishes to within 5° of the Sun at S-band. Appendix E provides the analysis. We therefore set the out-of-plane amplitude A_z to 140,000 km for a minimum elongation of 5.3°.

Minimum Drift Arc Duration (time between spin axis torquing maneuvers) must exceed 11-14 days. The tracking observables are so weak in L_1 orbits that long, unperturbed tracking arcs are needed. There was concern that thruster imbalance might cause coupling between axis torquing and translation along the orbit that would interfere with tracking. Although we feel imbalance can be adequately controlled, the issue is not closed, so the safer course is to guarantee a minimum of 11-14 days tracking unperturbed by thrusts. Also, planning and executing maneuvers more frequently than this could become impractical (expensive) in terms of ground support.

By means of computer simulation, we examined the relationship between minimum drift-arc duration, maximum Earth angle, and orbit size A_y. We constrained Sun angle to <20° and optimized each drift arc to try to center Earth angle within the antenna beam. We allowed Sun angle to go to zero and also tried constraining it to stay above 4-5°. The orbit was represented by an ellipse of semi-major axis A_y and semi-minor axis 140,000 km. X-axis motion was ignored. The period was set to 178 days, and we determined that starting phase made little difference.

Figure 4-3 shows Sun and Earth angles versus time for the orbit size we finally selected, with Sun angle held >4°. For comparison, Fig. 4-4 shows the same orbit, but with Sun angle permitted to go to zero. Figure 4-5 shows minimum drift arc durations for the complete range of choices examined, again with Sun angle permitted to go to zero. Also plotted in Fig. 4-5 is the minimum (worst-angle) transmit antenna gain for each Earth angle limit (a ±1° attitude-control error has been included).

Figure 4-5 clearly shows that the larger A_y orbits cannot meet all the constraints. Our RF downlink calculations indicated that about 12-14 dB of antenna gain was desirable, and the largest A_y that allowed us to meet all the other constraints was A_y=300,000 km. The cost of constraining the Sun angle to >4° was a decrease in minimum drift arc length from 14 to 11 days. We felt that was a worthwhile tradeoff to preserve our options for choice of attitude Sun sensor. The *SIMPL* orbit will therefore be an A_y=300,000 km "broken-Lissajous" approximation to a true halo, with orbit corrections needed every 1-3 months and spin-axis torquing maneuvers needed every 11-31 days. This represents a manageable level of ground support operations for SIMPL.

4.5 SOON/RSTN STATION VISIBILITIES

SIMPL's trajectory as viewed from Earth will approximate an ellipse centered on the Sun (Fig. 4-6). Theoretically, three optimally placed ground stations would be adequate to cover SIMPL, even with the additional 5.3° declination due to A_z. However, the station locations were not ours to select and are nonoptimally spread in longitude (two are nearly on a diameter). Four stations are therefore required.

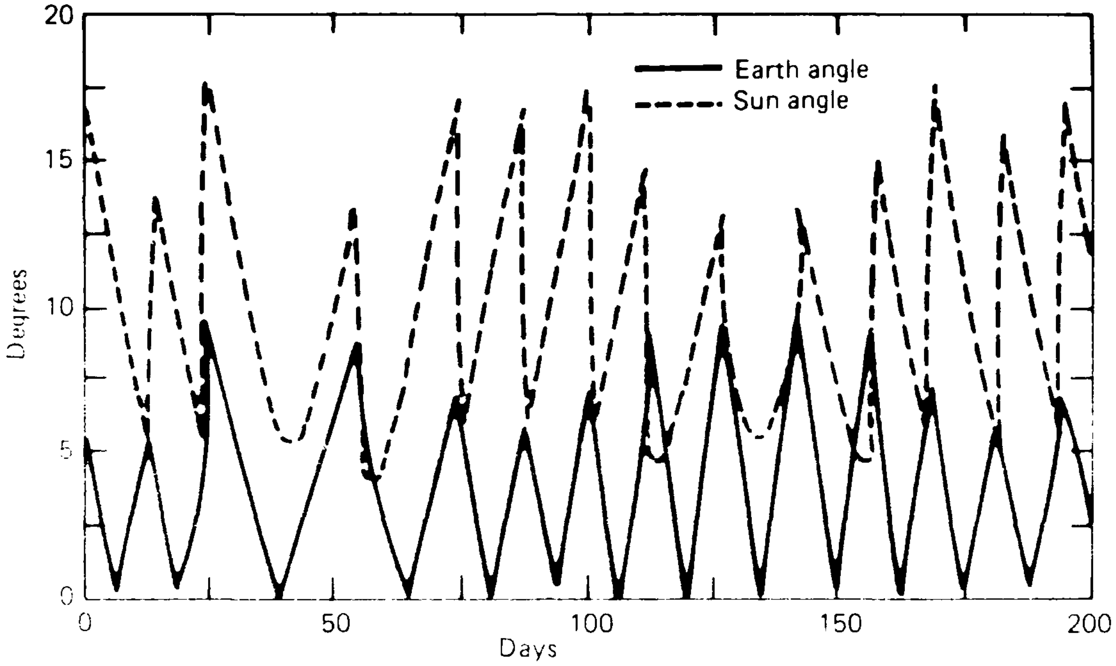


Fig. 4-3. Sun and Earth angles for A_y = 300,000 km orbit, $4^\circ <$ Sun angle $< 20^\circ$, Earth angle $< 10^\circ$. Min/avg/max drift arcs = 11.2/15.3/30.5 days.

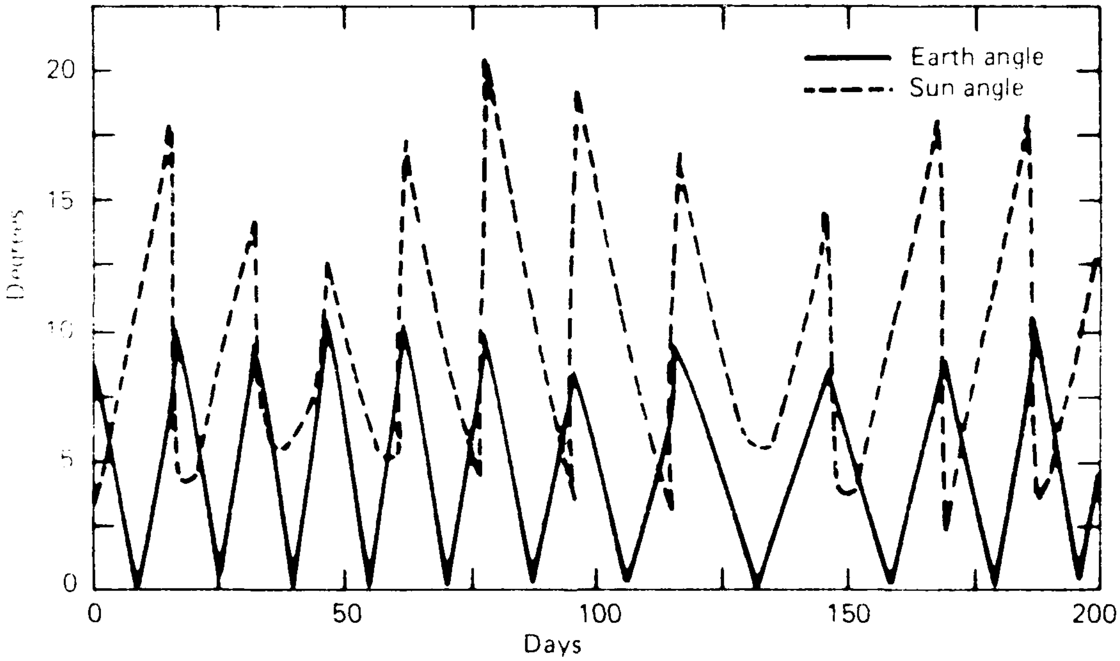


Fig. 4-4. Sun and Earth angles for A_y = 300,000 km orbit, no lower limit on Sun angle, Earth angle $< 10^\circ$. Min/avg/max drift arcs = 14/18/30 days.

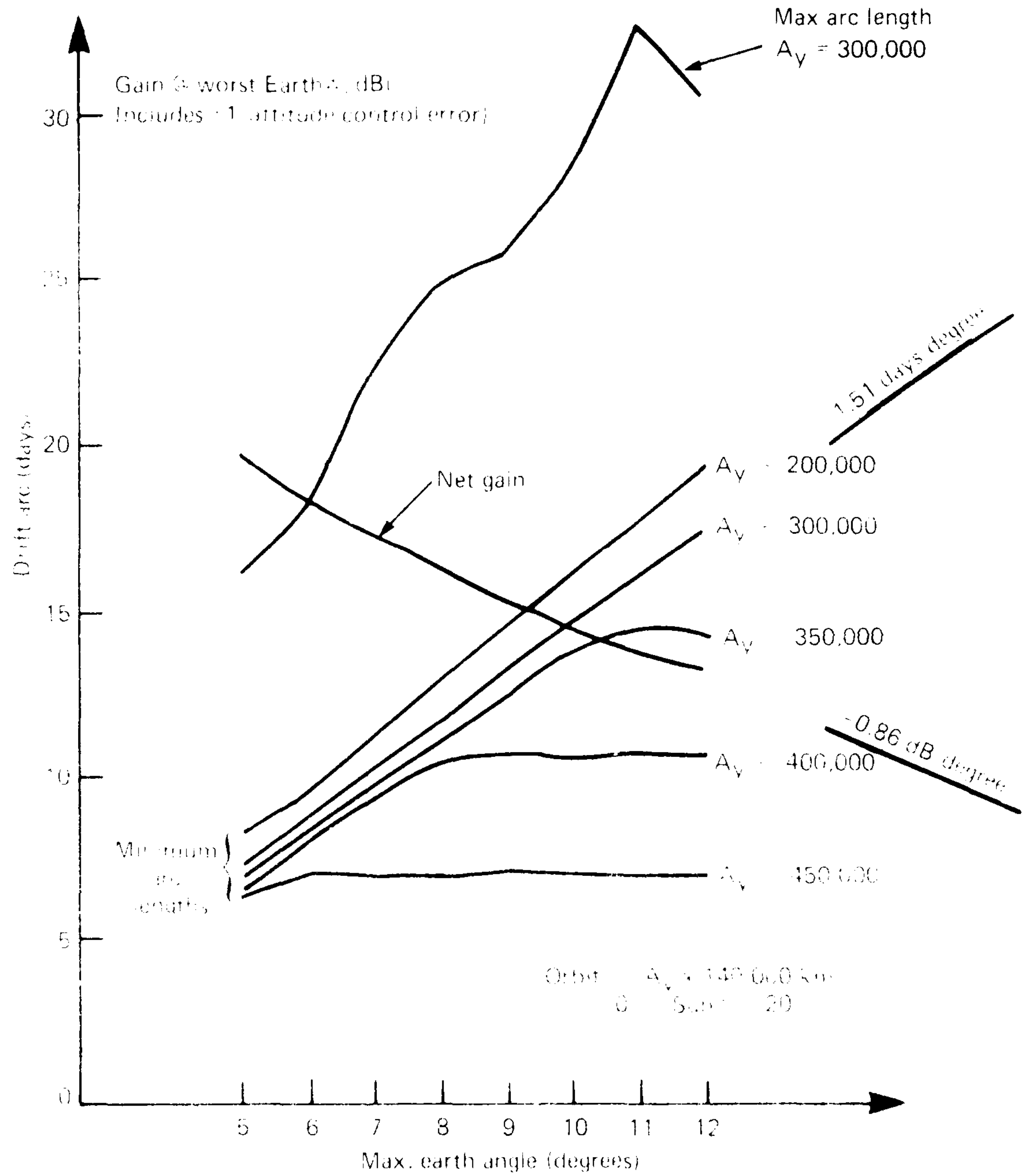


Fig. 4-5. Drift arc durations and worst-angle spacecraft antenna gain as a function of upper limit on Earth angle.

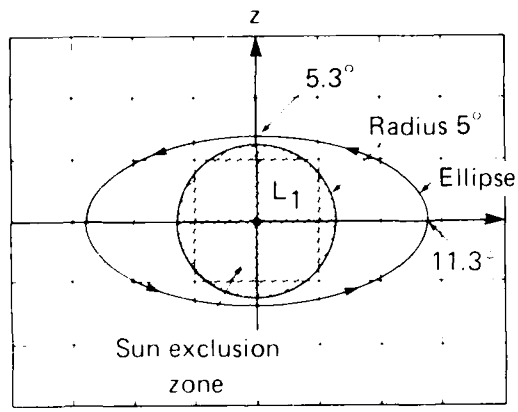


Fig. 4-6. SIMPL's trajectory as viewed from Earth (grid size 4° × 4°).

We computed SOON/RSTN station coverages under various conditions of Sun's declination and phase of SIMPL along its orbit. The effect on ground antenna elevations is the most important, since a minimum elevation of 5-10° is needed. A worst-case condition is shown in Fig. 4-7, in which the Sun is at its most southerly declination and SIMPL is at the most southerly portion of its orbit. Since three of the four SOON/RSTN stations are in the northern hemisphere, this represents worst-case coverage overlap. It also illustrates the worst-case maximum pass duration, at Australia.

From Fig. 4-7 we conclude that either Hawaii or California is acceptable for the fourth station, although Hawaii would provide better overlap. We also see that passes at Australia can be up to 13.5 hours long during southern-hemisphere summertime.

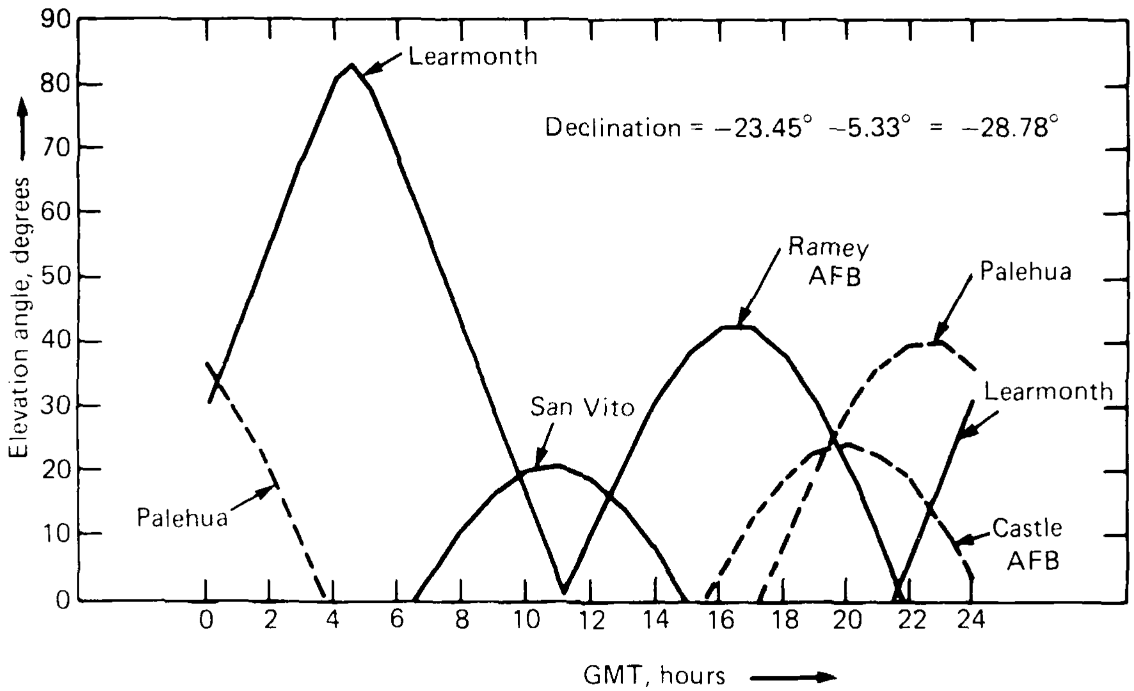


Fig. 4-7. **Worst-case elevation angles to "SIMPL" from candidate SOON/RSTN sites.**

4.6 DELTA-V BUDGET

Having selected orbit dimensions and spin scenario, a delta-v budget can now be established. This is a necessary input for the design of the on-board propulsion system.

The first requirement is to carry sufficient delta-v to correct PAM-D insertion errors. A conservative three-sigma allocation would be 150 m/s. For the insertion itself, 175 m/s must be carried, assuming a "slow" transfer orbit and no use of lunar-gravity assist. A "fast" transfer could save a month or so travel time to L_1, for an additional 80 m/s. Lunar assist could save about 80-125 m/s, depending on the complexity of the mission.

To perform the out-of-plane maneuvers for A_z=140,000 km, 24 m/s/year is budgeted. An additional 10 m/s/year covers orbit maintenance for A_y=300,000 km. The spin axis torquing maneuvers will require <1 m/s/year. Miscellaneous attitude maneuvers and margin are budgeted at 75 m/s. To summarize:

Correct launch errors	150 m/s
Orbit insertion	175 m/s
Orbit maintenance (6 yrs)	204 m/s
Spin-axis maneuvers (6 yrs)	6 m/s
Attitude and margin (6 yrs)	75 m/s
TOTAL	610 m/s required

In Section 6 we will show how low spacecraft weight permits this requirement to be easily met with the AMPTE-based design.

4.7 SPIN RATE

A high spin rate complicates the operation of star scanners, Sun sensors, and instruments. It raises the angular momentum which must be overcome during spin-axis changes. It can induce amplitude and frequency modulation on the omni antennas. Therefore the spin rate should be set no higher than necessary to satisfy attitude stability requirements, expel fuel from the tanks, and provide sufficiently rapid scanning by the instruments. We consider 5-10 rpm the range for operational spin rate and selected 5 rpm as our baseline.

5. LAUNCH SCENARIO

5.1 LAUNCH-VEHICLE OPTIONS

Although a Shuttle launch has been decreed for SIMPL, there is no technical reason why an expendable launch vehicle could not be used instead. Reasonable choices among expendables include the Delta, Atlas-E/F, and Thor. Each of these is easier to design to, and imposes less rigorous safety requirements, than the man-rated Shuttle. Expendables also, as a rule, provide more height inside the heat shield, which would be attractive if the spin-normal-to-ecliptic option had been selected and a tall antenna had been required. ISEE-3, a fairly tall spacecraft, was launched on a Delta 2914. Our AMPTE and GEOSAT spacecraft, from which SIMPL draws most of its heritage, were launched on a Delta 3924 and an Atlas-E, respectively. Subsystems were therefore designed to meet the mechanical, stress, and safety requirements of those vehicles.

5.1.1 Shuttle Constraints

Our design for NASA's WIND spacecraft was, like SIMPL, based on a Shuttle launch from Cape Canaveral. WIND was to use the McDonnell-Douglas PAM-D solid spinning upper stage to ascend from the Shuttle parking orbit to L_1. The PAM-D for WIND was to have the maximum propellant loading to increase loft weight, and use of the largest Sunshield was planned.

Because SIMPL's launch weight is little more than half of WIND's, we briefly examined the possibility of providing integral propulsion in place of the PAM-D. However, when launch support and Airborne Support Equipment (ASE) were factored in, there appeared to be no advantage in cost, schedule, or reliability.

The Shuttle has the advantages of increased payload size and weight, the possibility of crew interaction, and guaranteed launch support in SIMPL's time frame. The price paid is in meeting the rigorous safety requirements imposed on the payload because Shuttle is man-rated. The safety requirements and their design impacts are discussed individually in Section 6. They primarily impact the propulsion, mechanical, C&DH, and ASE designs and will show up as an indirect cost of launching on the Shuttle.

Based upon studies we did for the WIND spacecraft, it may be somewhat more difficult to protect delicate instrument sensors from contamination in the Orbiter cargo bay. After door closing, the environment is class 100 nominal, class 5000 guaranteed, with 15 ppm HC contamination. JHU/APL has discussed with JSC and KSC representatives improving this performance for contamination sensitive instruments. We have explored extra cleaning and inspection steps, bay liners, instrument covers, and both local and whole bay purging. The most sensitive sensors on SIMPL will be in the Solar Wind and Particle instruments. We do not expect these to be as sensitive to contamination as WIND's were.

Another constraint, which turned out not to apply to SIMPL, is that only 100 inches of height is available for spacecraft that must sit atop the PAM-D. In fact, our SIMPL design is so compact that Shuttle size and weight limitations became irrelevant. We were not only able to fit SIMPL into the smallest available PAM-D Sunshield, thereby saving substantial launch costs, we even have height and loft weight capability available for a subsatellite.

5.1.2 PAM-D Operations

Figure 5-1 shows the PAM-D system hardware used for Shuttle launches. A cradle assembly supports the combined PAM-D/payload, spin table, PAM-D and spacecraft ASE, and foldable Sunshield (not shown). All this equipment, except the PAM-D and payload, is returned to Earth for re-use.

The PAM-D itself consists of a Thiokol STAR-48B solid rocket motor, a payload attach fitting with 37-in. bolt hole circle, and electronic equipment. An S-band telemetry system is included to transmit vehicle performance data to the Orbiter or to the ground. Nutation damping is not provided by the PAM-D; the spacecraft must provide this function. We have included a conservative 150 m/s in our propulsion budget to correct worst-case PAM-D insertion errors.

Prior to deployment from the Orbiter cargo bay, the payload is supported by its own ASE, interfacing to the spacecraft via slip rings, and to the Orbiter avionics through the PAM-D ASE (see Section 6.11 for ASE details). The folding Sunshield protects the payload from thermal extremes until just a few minutes before launch. The spin table imparts a stabilizing spin (40-100 rpm) to the PAM-D/payload combination prior to deployment. Deployment occurs when redundant bolt cutters release a vee-block clamp band. Springs then eject the spinning PAM-D/payload out of the cargo bay at >2.5 fps. Shuttle safety rules require that the payload drift for >45 minutes before the PAM-D is ignited.

We asked McDonnell-Douglas to compute the loft weights and other parameters for a mission to L_1 using the most recent PAM-D performance data. We assumed that 3200 m/s of additional

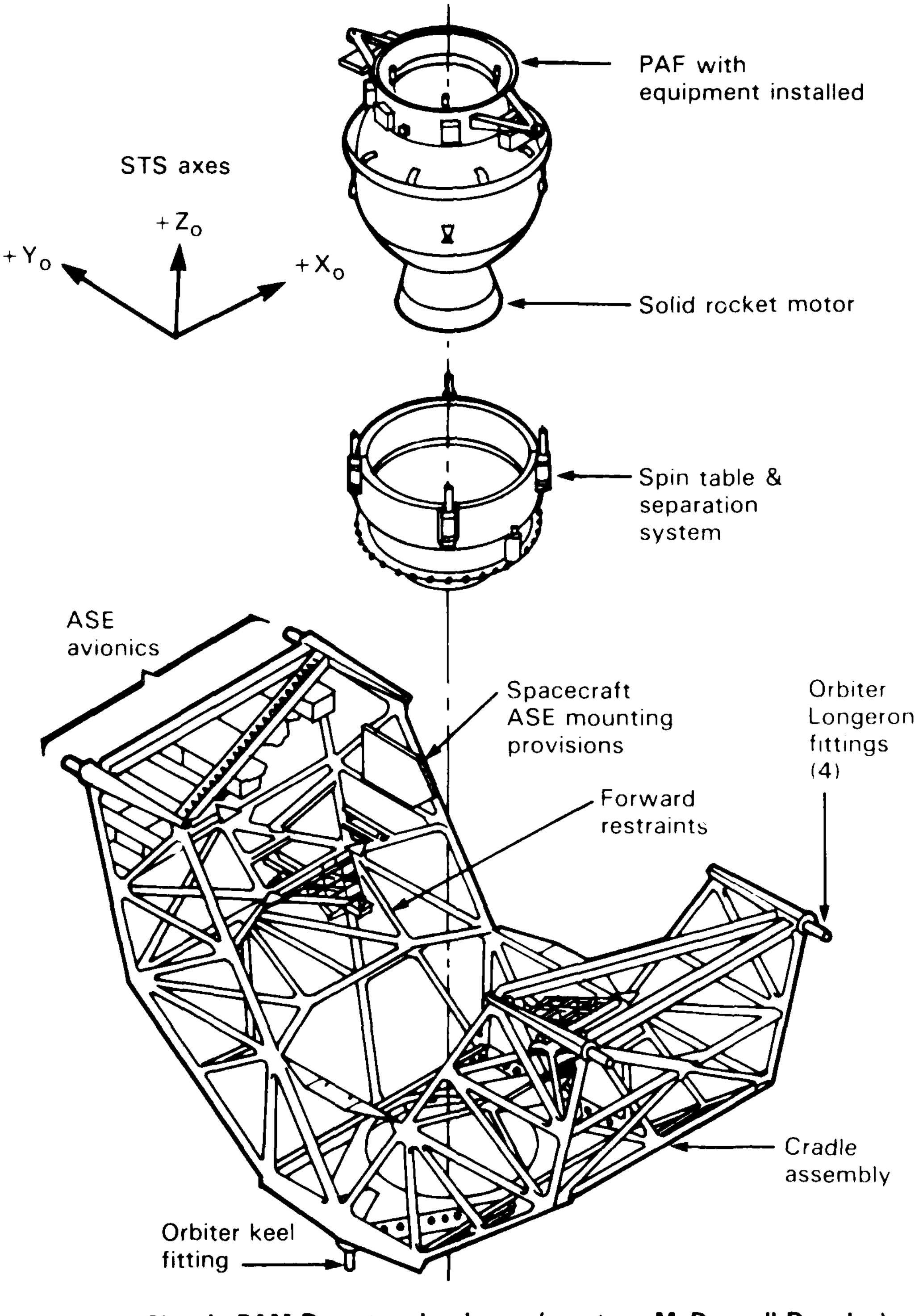


Fig. 5-1. Shuttle PAM-D system hardware (courtesy McDonnell Douglas).

delta-v would be needed from the PAM-D, vectorially added to the 7730 m/s from the Shuttle parking orbit. Figures were computed for both an off-loaded ("baseline") PAM-D in the small (86-in.) Sunshield and the maximally loaded PAM-D in the large (115-in.) Sunshield. The results are summarized below.

	Baseline PAM-D Small Sunshield	Maximum PAM-D Large Sunshield
Inside diameter	86 in.	115 in.
Length along bay	93 in.	136 in.
Propellant loading	1755 kg	2024 kg
Allowable spacecraft wt.	608 kg	747 kg
Total element weight	3668 kg (8088 lb)	4066 kg (8966 lb)

As will be seen in Section 6, SIMPL will weigh only 393 kg, even including 20% growth factors. Therefore, not only is the small Sunshield sufficient, there is substantial excess loft capability that must be dealt with.

5.1.3 Subsatellite and Other Options

There are a numer of options for dealing with the excess PAM-D capability. These include:

1. offload additional propellant from the STAR-48B motor;

2. substitute a smaller, previously Shuttle qualified rocket motor, possibly the STAR-37;

3. use an intentionally inefficient trajectory (i.e., trade extra delta-v for additional launch window flexibility);

4. add ballast to the PAM-D (<u>not</u> to the satellite, as that would reduce delta-v during the mission); or

5. carry a <u>subsatellite</u>.

By far the most interesting option is the <u>subsatellite</u>. Based on the 608 kg loft capability, the subsatellite could weigh about 200 kg, and (see launch configuration in Section 6.6) could be about 20-30-in. high. It would have to mate to 37-in. bolt hole circles at each end. Some additional structural strengthening might be required due to the longer cantilever. The subsatellite would be placed in a highly elliptic orbit; it might need to carry some propulsion of its own to modify that orbit.

5.1.4 Launch-Cost Factors

Shuttle launch costs are determined by the "length factor" or the "weight factor," whichever is greater. Length factor is the length along the bay, rounded up to the next inch, divided by the bay length of 60 ft. Weight factor is the total element weight divided by the 65,000 lb Orbiter payload capability. For the small and large PAM-Ds for SIMPL's orbit, the factors are:

	Baseline PAM-D Small Sunshield	Maximum PAM-D Large Sunshield
Length factor	0.129	0.189
Weight factor	0.124	0.138

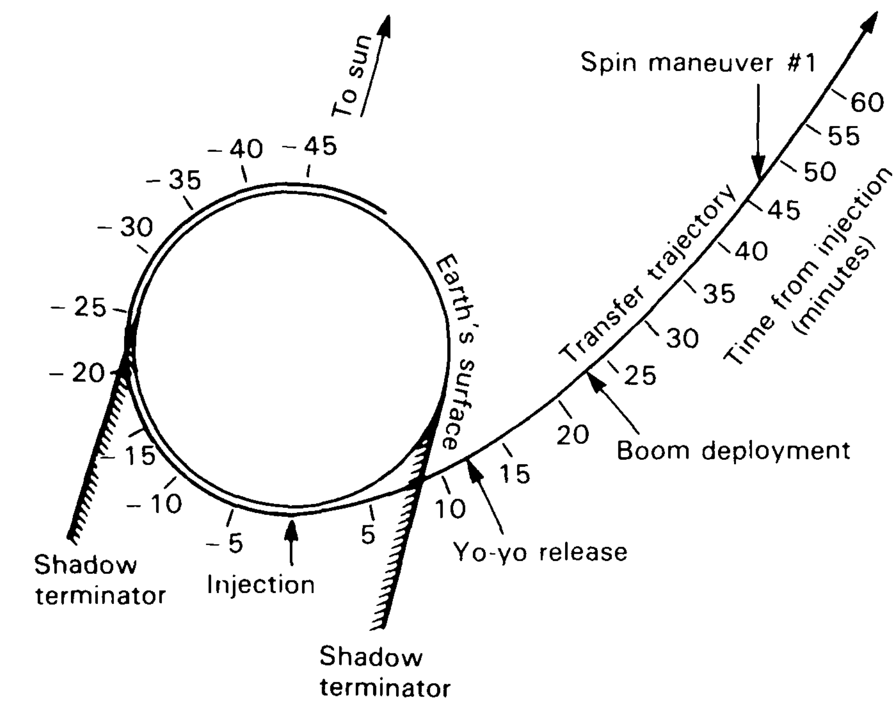
Thus, in either case, SIMPL will pay by length along the bay. Using the standard NASA manifesting factor of 4/3 and estimating the DoD launch cost in SIMPL's time frame (now being negotiated), we see that by fitting SIMPL into the small Sunshield, we will save the Air Force about $5.5 million in NASA launch costs.

5.2 TRAJECTORY AND LAUNCH

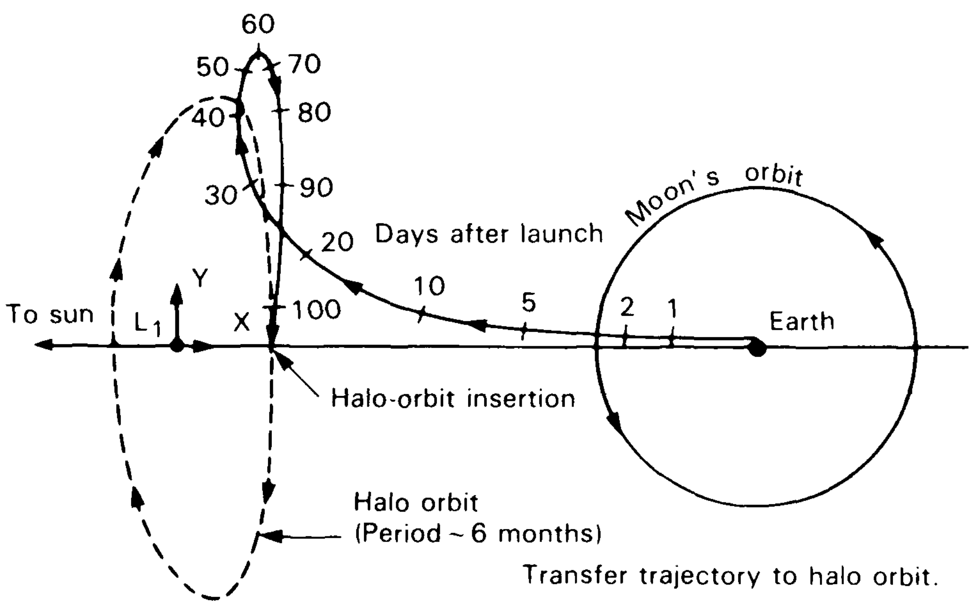
5.2.1 Transfer Trajectory and Launch Windows

There is no time-of-year launch constraint for SIMPL. However, for optimal (lowest energy) transfer to L_1, insertion should take place along a path inclined about 5° to the ecliptic. Therefore, for a given time of year, there is an optimum time of day of launch, yielding the optimum ascending node. The launch window is about five minutes or so for a delta-v cost of 10 m/s. In general, this will <u>not</u> be the noon/midnight GMT ±1 hour orbit frequently used by Shuttle. Time of deployment from the Shuttle determines argument of perigee.

Figure 5-2, based on the ISEE-3 launch (Ref. 5-1), shows the ideal case parking orbit and early transfer phase for travel to L_1 from the 296 km, 28.5° Shuttle parking orbit. The optimum PAM-D ignition point is about 14° past the local midnight point. Because of the 45 minute drift period required by Shuttle safety rules, deployment from the Orbiter must occur at approximately the



Parking orbit and early transfer phase.



Transfer trajectory to halo orbit.

Fig. 5-2. SIMPL launch scenario (after Farquhar, Ref. 5-1).

local noon point. The 37 minutes spent in eclipse is preceded by about 25 minutes with SIMPL's solar panels folded down. This sets the battery requirement, discussed in Section 6.3.

When an expendable launch vehicle is used (as for ISEE-3), this ideal case can be approximated closely. A Shuttle launch, however, can carry four or more spacecraft, none of which are likely to have the same ascending node requirement as SIMPL. That means a compromise parking orbit will have to be selected, and early transfer operations may get more complicated. In an extreme case, it may be necessary for SIMPL to enter a "phasing orbit" to compensate for non-ideal launch conditions. This Shuttle manifesting problem affects WIND, too, and NASA/GSFC will soon begin studying solutions. The results of that study can be applied to SIMPL as well.

Our propulsion delta-v budget has assumed a conservative direct ascent to L_1, but lunar swingby gravity assist could possibly save about 80 m/s delta-v. If lunar swingby were later elected, a phasing orbit might then be desirable to avoid a time of lunar month launch constraint. There is no other need for lunar phasing since SIMPL, unlike ISEE-3, will not use the moon as an attitude reference at any point in the mission (See Section 6.4).

The lower portion of Fig. 5-2 shows the later transfer trajectory. An energy saving "slow" transfer trajectory to L_1 is shown. By day 25, SIMPL has essentially arrived at L_1. The remaining time is spent in the insertion maneuver, with SIMPL actually achieving halo orbit at about day 118.

5.2.2 Launch Timeline

Table 5-1 shows milestones in the launch of SIMPL, beginning with the delivery to KSC about three months prior to liftoff. Following checkout of the spacecraft, SIMPL will be mated to the PAM-D. The entire manifest will then be assembled in the Vertical Processing Facility and installed in the Orbiter cargo bay. Liftoff is shown about two days prior to desired deployment from the Orbiter. About an hour before deployment PAM-D power is activated and the Orbiter maneuvers to deployment attitude. At T - 14 minutes the Sunshield is opened and spin-up begins. At T = 0 deployment takes place. Once the payload has separated from the Orbiter by >200 feet, active nutation thrusting can begin. The PAM-D/SIMPL then drifts for the required 45 minute safety period.

At T + 45 minutes the PAM-D ignites, burning out about 80 seconds later. The PAM-D is jettisoned as soon as possible following burnout and SIMPL is turned to maintain the Sun angle <20°. The solar panels are erected and the booms and wires deployed as soon as possible, so that minimum time is spent in the unstable moment of inertia configuration. Deploying these appendages despins the spacecraft to its operational spin rate, as trimmed by the spin-control thrusters. The active nutation control system can then be turned off. At about T + 1 day the first mid-course correction is performed.

SIMPL now heads toward L_1 along a trajectory minimizing insertion delta-v and holding solar elongation >5° so that ground contact can be maintained. Attitude is held at a compromise angle to satisfy power and thermal constraints, yet still permit switch-over to the high-gain spacecraft antenna as slant range builds up. SIMPL inserts at around T + 118 days as described above.

Table 5-1

SIMPL launch timeline

t - 3 months	Deliver to KSC, mate to PAM-D
t - 2 days	Shuttle liftoff
t - 14 minutes	Open Sunshield, begin spin-up
t = 0	Deploy from Shuttle, ≥ 1 fps, 60 rpm Begin active nutation control @ d > 200' Drift 45 minutes
t + 45 minutes	Ignite PAM-D
t + 46.5 minutes	PAM burnout, jettison PAM Sun angle $\rightarrow$ $0° \pm 20°$, elongation > 5° Erect panels (despin) Deploy booms, wires (further despin) "Slow trajectory" to L_1
t + 1 day	First mid-course correction
t + 25 days	Arrive vicinity of L_1
t + 118 days	Insert into L_1

6. SUPPORT SPACECRAFT

6.1 SYSTEM DESCRIPTION

We considered two JHU/APL spacecraft designs as the basis for SIMPL -- AMPTE and WIND. The AMPTE/CCE was designed and built by JHU/APL and successfully launched in August 1984. A brief description of AMPTE is given in Appendix A. WIND was carried through conceptual design by JHU/APL for NASA, and was entering into preliminary design when NASA's ISTP program was delayed one year.

Our WIND design consisted of an 8 ft diameter, 4 ft high body carrying seven complex instruments and 10 appendages. It was designed for spin normal to the ecliptic, a NASA science requirement. When ISTP cost reductions were being sought, JHU/APL explored a version of WIND modified to operate with the spin axis parallel to the ecliptic, like SIMPL. Although WIND would be more than ample for SIMPL's requirements, we felt that if a design could be based on our AMPTE bus, that would yield a lower cost.

We therefore began by attempting to fit SIMPL onto the AMPTE bus. As it turned out, this approach succeeded and we did not have to fall back to the WIND design.

AMPTE has the advantage of heritage and of being designed from the start for Sun-facing operation. Its light weight minimizes the amount of propellant that must be carried to achieve SIMPL's mission delta-v. Its compact size imposes a natural restraint against adding features not absolutely needed to meet SIMPL's operational goals. This "creeping expansionism" is often

a source of cost growth on large spacecraft. AMPTE's light weight also increases launch flexibility and opens the opportunity for a subsatellite.

AMPTE was designed to mate to one of the spacecraft below it on the stack, so an adapter section was required to mate to the larger PAM-D bolt hole circle. We elected to make this section tall enough to house several of the SIMPL subsystems, such as the RF components and star scanners. The space between the narrow end of the conical adapter and the bottom of the AMPTE main body provided a natural home for the large propulsion system required by SIMPL. Heat from the RF components could keep the tanks warm, while the solar arrays could shade the tanks from the Sun. The frontispiece illustrates the resulting SIMPL orbit configuration (multilayer insulation deleted for clarity).

Many of the items flown on AMPTE could be removed to make room for SIMPL's subsystems. Major deletions included:

- the five instruments
- the STAR-13A solid rocket motor (SRM)
- safe-and-arm and thermal shielding for the SRM
- one battery
- the magnetic torquing system
- the tape recorder and its radiation shield
- the cold gas system
- all (or all but one) of the thermal louvers

We found that removal of these items, and the addition of the conical adapter section, yielded more than enough room for SIMPL's components (see Section 6.6). Table 6-1 compares the original AMPTE/CCE and our SIMPL design.

Table 6-1

SIMPL vs. AMPTE/CCE

	SIMPL	AMPTE/CCE
Spin axis	point ~ to Earth	point to Sun
Spin rate	5 rpm	10 rpm
Attitude detection	SS + SSSS	SDSAD + mag.
Attitude control	hydrazine	cold gas + z coil
Launch vehicle	Shuttle/PAM-D	Delta
Mates with	PAM-D	CCE/IRM adapter
On-board propulsion	hydrazine (800 m/s)	solid (600 m/s)
Size	4 ft. sq. x TBD	4 ft. sq. x 1.3 ft. high
Number of deployables	4	1
Experiment wt. (incl. proc.)	34 kg	32 kg
Experiment power (incl. proc.)	20 W	27 W
Total S/C weight	393 kg	242 kg
Solar array output	225 W	140 W
Average power load	122 W	95 W
Main power bus	reg. 28 V	unreg. 28 V
Battery	1 x 6 A-H	2 X 4 A-H
Average data rate	100 bps	3300 bps
Radiation environment	moderate	high

6.1.1 Spacecraft Heritage

Although our SIMPL design is based heavily on AMPTE, there is considerable heritage from other programs as well. From AMPTE, we have taken the structure, array design, thermal design, Sun sensor, hemispherical coverage antennas, and GSE. From our GEOSAT-A comes the concept for fully redundant C&DH, the COMSEC approach, and experience with small ground stations. SIMPL's magnetometer boom is a subset of the one used on our MAGSAT. From our WIND design comes the propulsion and power system concept, and the Shuttle/PAM-D interface. WIND's power system concept, in turn, was derived from ISEE-3. The instruments, too, will draw heavily from prior programs: the Magnetometer from AMPTE, the Solar Wind instrument from either Ulysses or Voyager, the Particle instrument from IMP-8, and the Radio Wave instrument and wire booms from ISEE-3.

6.1.2 Power and Weight Estimates

SIMPL's power and weight breakdown by subsystem is shown in Table 6-2; (a more detailed breakdown to the black box level is given in Appendix C). SIMPL will require about 122 W orbit average power and will have a wet weight at launch of 393 kg. As would be expected in this high delta-v mission, the propulsion system is the dominant weight item. The instruments, their processors, and the communications equipment account for another large fraction of the resources. At this early stage of design, we feel it is advisable to carry generous growth allowances. A conservative 20% growth is carried on all weights and powers except for the hydrazine propellant (the tanks are already filled to maximum).

Table 6-2

SIMPL power and weight summary

	Orbit Average Power (W)	Weight (kg)
Instruments, Processor, Booms	21.8	44.3
Attitude System	3.5	12.0
Propulsion System	---	141.5
Power System	12.9	26.5
Communications	52.3	41.9
Thermal Control	11.0	4.4
Miscellaneous	---	32.0
Structure	---	45.4
20% Growth Allowance	20.3	45.0
TOTALS	122 W	393 kg

6.2 REDUNDANCY PHILOSOPHY

The spacecraft lifetime goal was given as six years, with four years minimum. Associated probabilities of success were not given. SIMPL was not required to eliminate all single-point failures, but we have attempted to minimize them.

As an operational mission, SIMPL requires an increased level of redundancy beyond that of AMPTE. We have incorporated full redundancy in almost every system, unless its use imposed undue burdens on the design or the risk of failure was small. Almost all electronic systems, for example, are redundant. This includes the power, attitude control, C&DH, instrument data processor, and RF systems. Although only a single battery is carried, SIMPL is designed to operate without a battery (solar-only mode) once at L_1. The propulsion and attitude detection systems are fully redundant. The two most important instruments, the Magnetometer and the Solar Wind Plasma Instrument, are at least partially redundant.

We have also enhanced reliability by minimizing deployed items. Of the five communications antennas, for example, none are deployed.

However, any practical spacecraft design contains some single point failure modes. For example, loss of the high-gain antenna would prevent operation to the SOON/RSTN stations. Failure of the boom to deploy would compromise magnetometer data. These are examples of subsystems having inherent simplicity and reliability, while at the same time being impractical to duplicate on a small spacecraft.

Figure 6-1 is the SIMPL spacecraft block diagram. Sections 6.3 through 6.11 describe the SIMPL subsystems in more detail.

6.3 POWER SYSTEM

The SIMPL power system has been designed with the objective of minimizing both conducted and radiated electromagnetic interference, reducing static charge accumulation, and providing functional redundancy throughout. (Specific features are listed in Table 6-3.) It is a type of DET (Direct Energy Transfer) system that was used on ISEE-3 and a number of other NASA/GSFC spacecraft. It features a charge control system of low power dissipation and is designed to function without the battery ("solar only" operation). Solar-only backup is particularly suitable for this mission with its unique 100% Sun condition, allowing continuous operation even if the battery fails. This provides a backup for battery failure without the additional weight, volume, and magnetic disturbance that would result from the use of a second battery.

6.3.1 Solar Arrays

6.3.1.1 Radiation Damage Considerations

Radiation damage calculations are based on data from Appendix B. Table 6-4 shows a pessimistic calculation of the net fluence reaching the surface of a solar cell as a function of coverglass thickness at the end of a six-year lifetime. To show the effects on solar cell performance, these data are superimposed on the plot of Fig. 6-2. The solar cell performance degradation

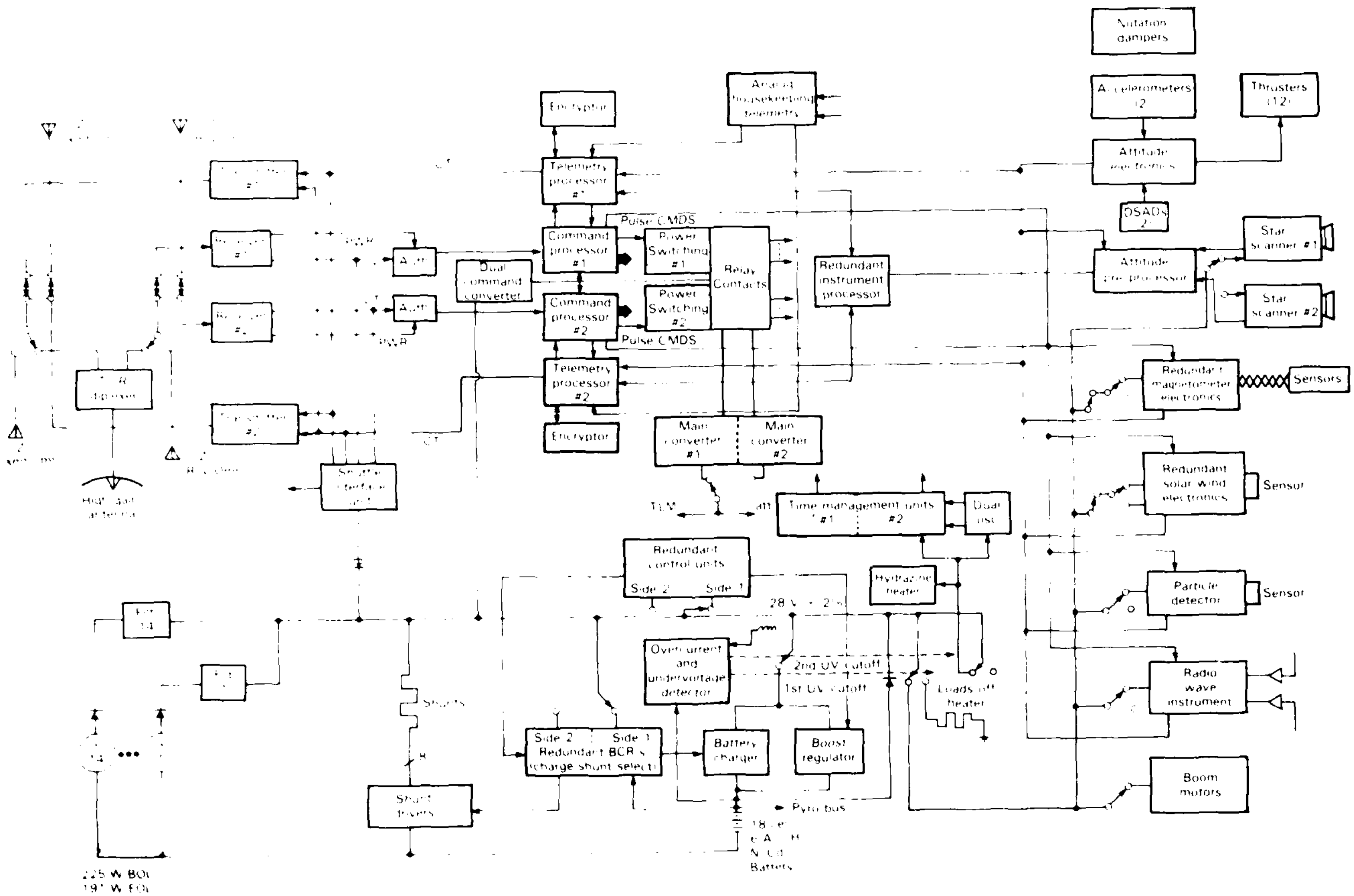


Fig. 6-1. SIMPL spacecraft block diagram.

Table 6-3

SIMPL general power system features

Four year minimum life.

Six year design goal.

28V ± 2% regulated main power bus.

Capable of operating without battery (in "solar only" mode).

20% EOL margin in load summary.

55% additional margin in array design.

Coplanar, unshaded solar panels.

Virtually no solar array current variation.

Solar cell strings & wiring to magnetically cancel.

Conductively coated coverglass on the solar cells to reduce static charge accumulation.

Solar cell substrates use conductive thermal control coatings to reduce static charge accumulation.

Maximum battery depth of discharge = TBD (see text)

Undervoltage limit provided.

Overcurrent limit provided.

Maximum bus ripple = 200 mV p-p (1 Hz - 100 kHz).

Redundant essential loads to be fused.

Ordnance bus capability: 50 A for 50 ms at 16 A minimum.

Battery can be re-conditioned in orbit.

Battery will support:

1. Transfer orbit needs.
2. Pyrotechnics
3. Eclipse loads (needed only early in mission)

Any necessary converters will be designed at high frequencies minimize their interference with instruments.

Table 6-4

Equivalent 1 Mev electron fluence as a function of
solar cell coverglass thickness

Coverglass Thickness (mils)*	Total No. of Equivalent 1 Mev Electrons in a Six-Year Lifetime	Solar Panel Wt (lb)	Weight Per EOL Watt (lb/watt)
20	3.96×10^{14}	25.0	0.132
12	5.06×10^{14}	23.4	0.125
6	5.34×10^{14}	22.4	0.120

* 6 mil cover is Microsheet Glass. All others are Corning 7940 fused silica.

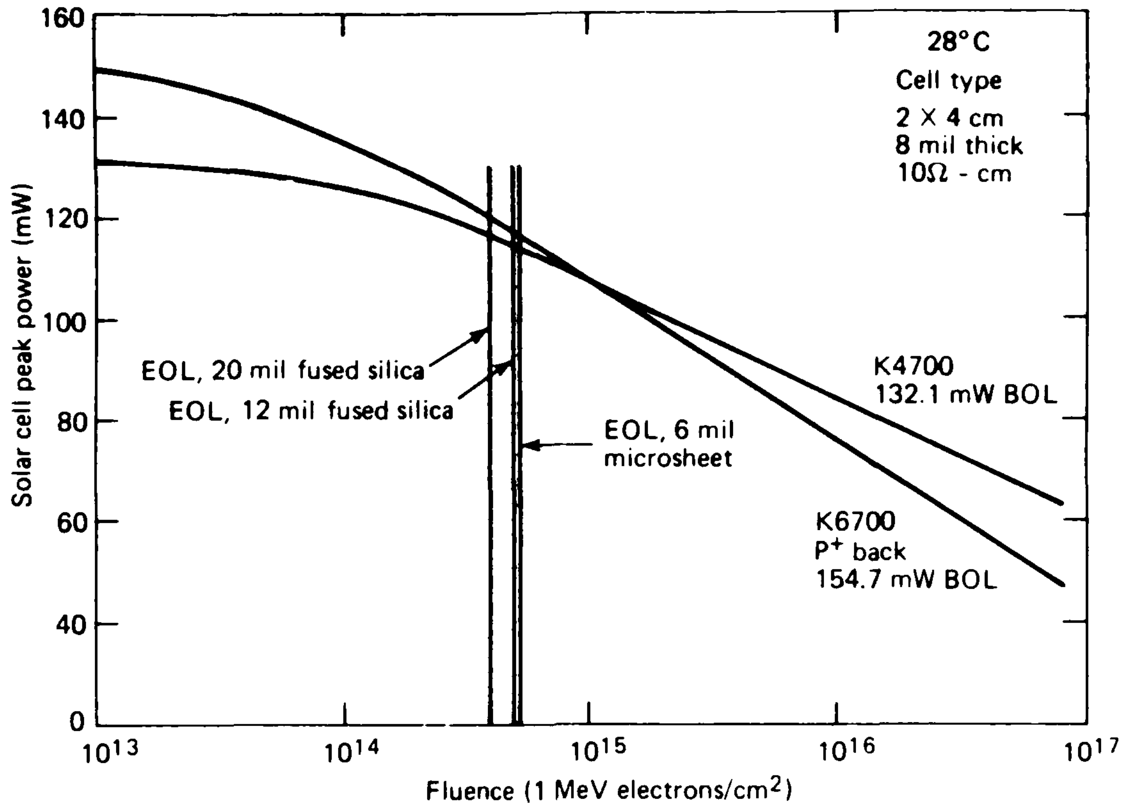


Fig. 6-2. Solar cell output versus radiation (end of six year life at L_1 shown for SIMPL).

curve represents the loss in peak power as a function of accumulated radiation for the two most likely cells to be used for this application. Clearly, the K6700 cell with the P^+ backing has the largest initial output, even though it degrades faster with radiation exposure. It is still the best choice for a satellite like SIMPL with relatively low fluence.

Table 6-4 also shows the effect of the coverglass thickness on the solar panel weight and, more importantly, its weight per end-of-life watt. The six-mil glass would be the best choice if weight were the only consideration. However, it is desirable that the coverglasses on this spacecraft be coated with a transparent, conductive material, such as indium oxide, and be electrically interconnected and grounded to the structure. Fulfilling this requirement may be more difficult on the thinner,

more fragile, six-mil thick coverglasses. Therefore we are presently proposing to use the less delicate 12-mil thick, fused-silica covers.

6.3.1.2 Solar Cell Array Configuration

Figure 6-3 shows one of the four solar cell panels with a solar cell layout. The 31.1 inch dimension is unchanged from the AMPTE spacecraft, while the 23.97 inch dimension represents a 16% increase over what was used on AMPTE. The area of each of the four SIMPL panels is 5.2 ft^2.

There are 74 2 x 4 cm solar cells in series and seven cells in parallel on each of the four panels. Power return wires will be run under the solar cells so as to cancel the resulting magnetic field. All solar cell covers and the back side of the solar panel will be covered with conductive coatings and connected to spacecraft ground in order to reduce static charge accumulation.

6.3.1.3 Solar Cell Array Performance

Average solar cell I-V curves are shown in Fig. 6-4 for the beginning and end of life and for the maximum operating and standard (reference) temperatures. The range of operating point, determined by the bus voltage variation, is superimposed to show the solar cell current (or power) available. At beginning of life we have 306 mA (119 mW) per solar cell and at end of life we have approximately 258 mA (101 mW) per cell.

Figure 6-5 shows the power available from the solar cell array as a function of angle to the Sun for beginning and end of life. Since the array is coplanar, these curves represent both

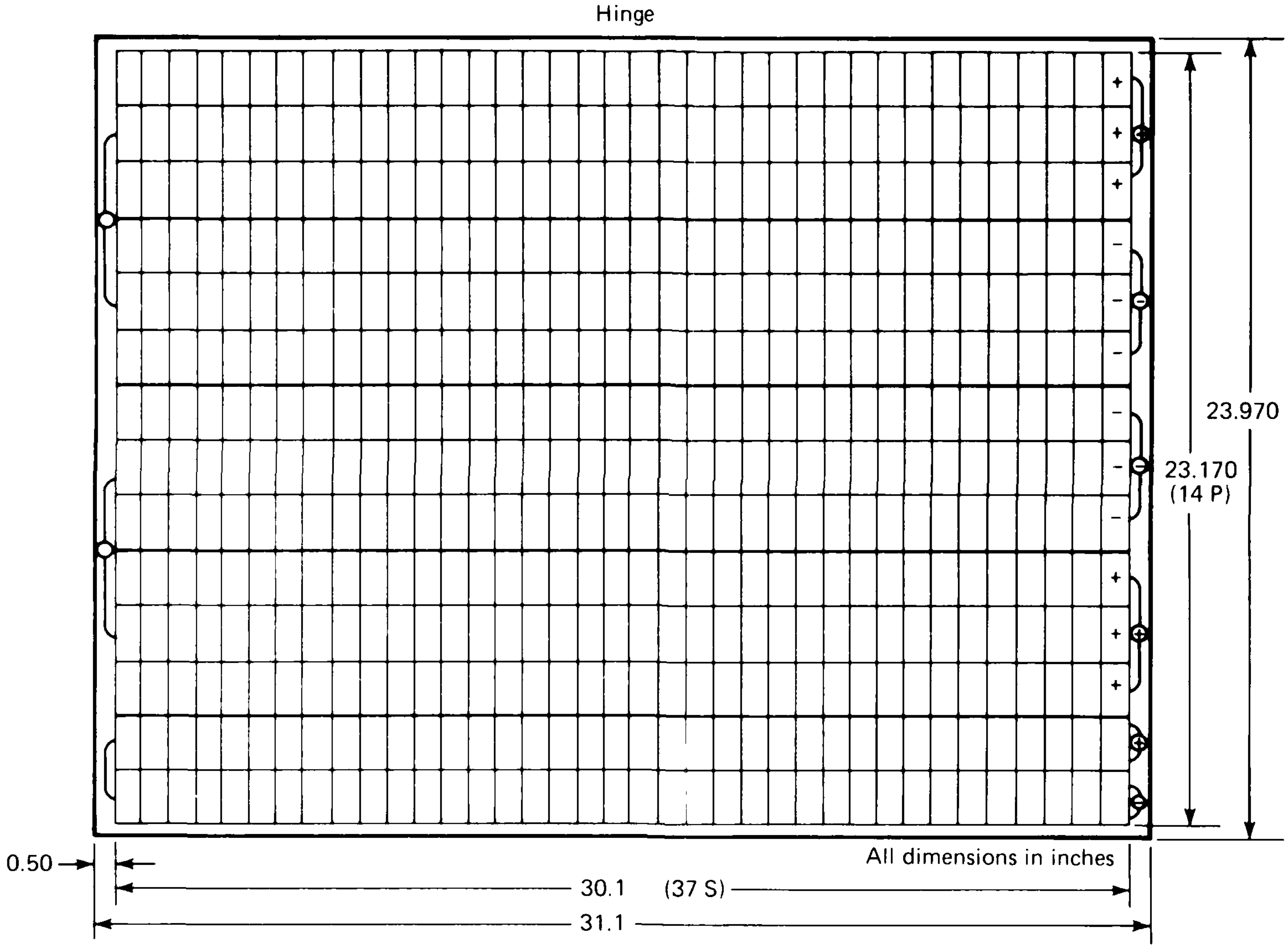


Fig. 6-3. SIMPL solar panel layout.

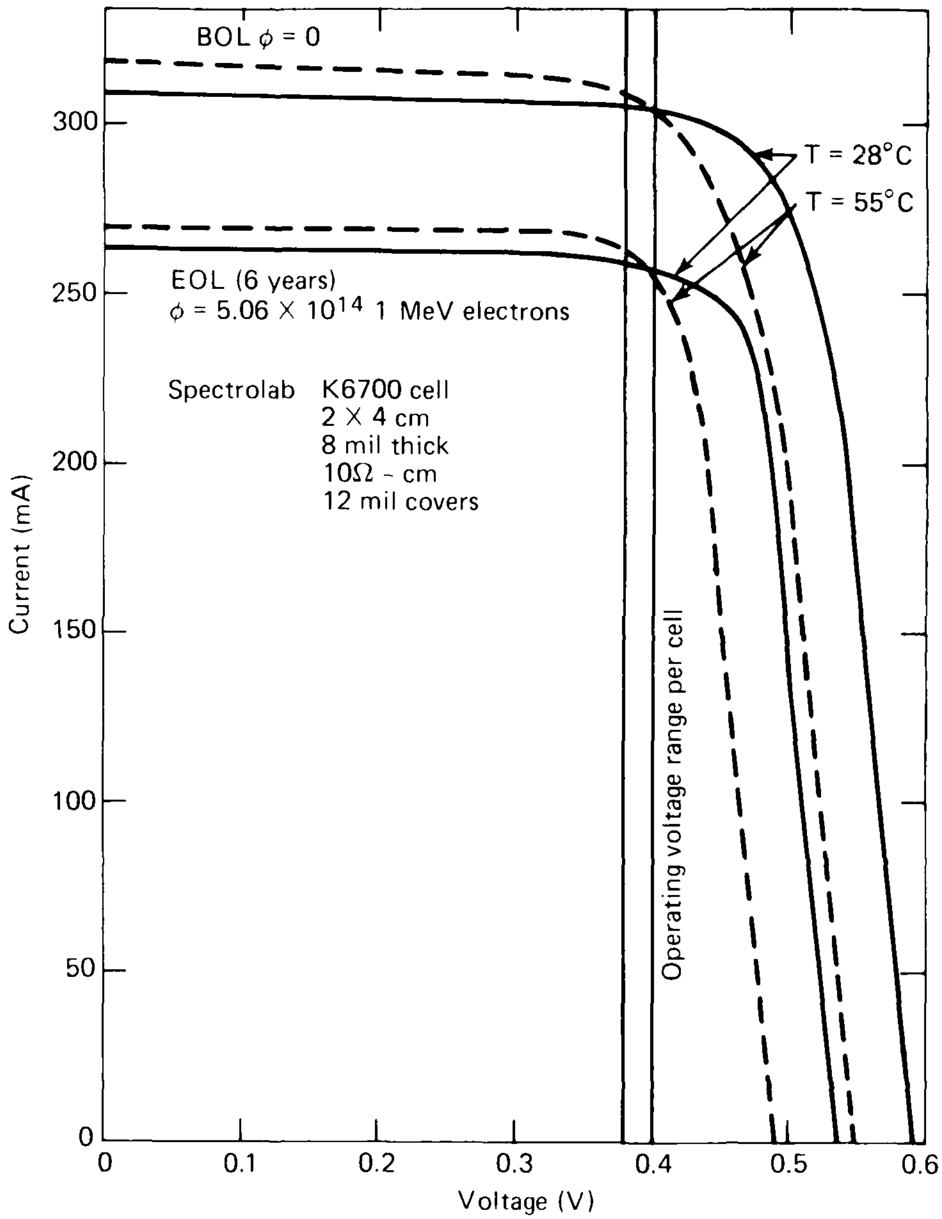


Fig. 6-4. SIMPL solar cell I-V curves using conductive coverglass of 90% transmission.

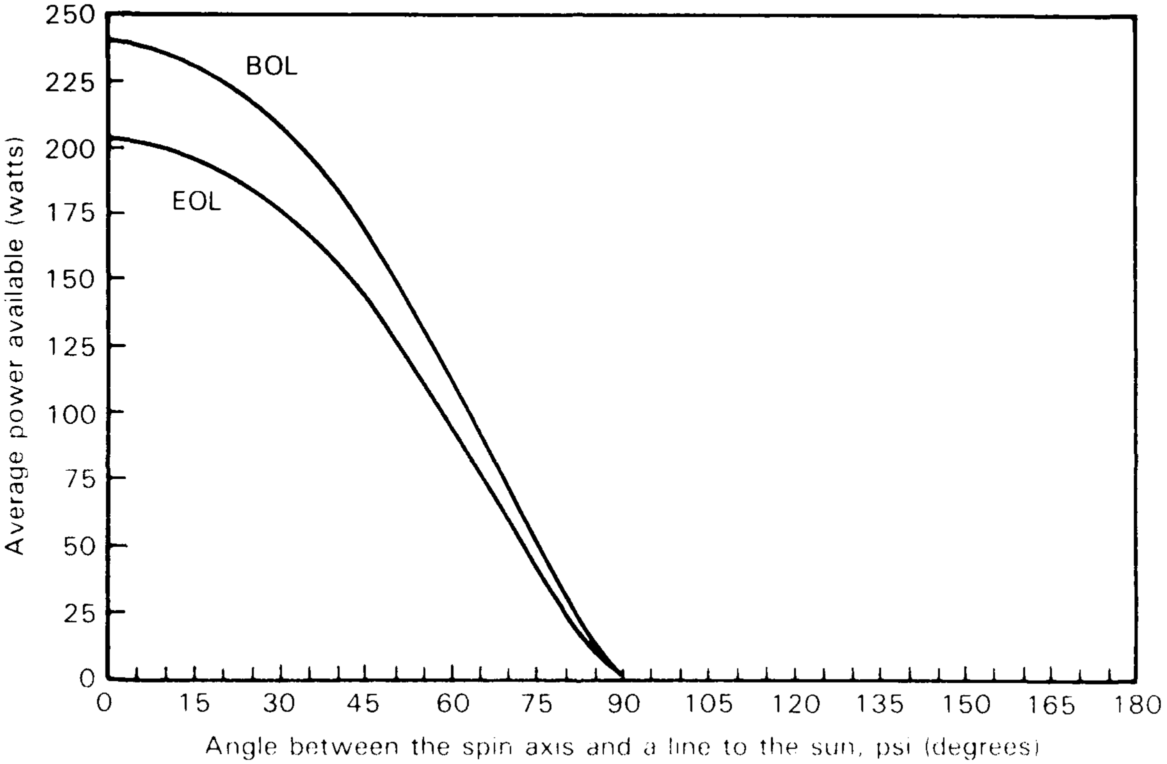


Fig. 6-5. **SIMPL average array power versus angle psi.**

the instantaneous and long term average current. Psi is the angle between a line to the Sun and the +Z axis of the spacecraft (which is coincident with the solar array normal vector). Therefore, at psi = 0° the available array power is 240 watts BOL and 202 watts EOL. (This represents 11.6 watts per ft^2 BOL and 9.75 watts per ft^2 EOL.) At the predicted maximum Sun angle of psi = 20°, the available power is 225 watts BOL and 191 watts EOL.

6.3.2 Battery

Only one battery will be needed on SIMPL since, after it is on station, the spacecraft could operate continuously in the "solar only" mode if required due to a battery failure.

The battery size is determined entirely by the requirements of the transfer orbit, which are not precisely defined at this time. It is established that, after separation from the Shuttle, the spacecraft will drift while attached to the PAM-D for a period of about 45 minutes. Also, during and after the drift period the spacecraft will be in a 300 km, circular orbit at 28.5° inclination. This type of orbit has a maximum eclipse time of 37 minutes (See Fig. 6-6).

An optimistic scenario, resulting in minimum battery demand (45 minutes), would result from a complete overlap of the drift and eclipse periods. A pessimistic scenario, resulting in maximum battery usage, would occur if the maximum (37 minute) eclipse period were to immediately follow the 45 minute drift period. This would result in a total discharge period of 82 minutes. In each of the above cases, there would actually be some small amount of solar array current due to the Sun incident on the folded solar panels during at least part of the drift period. For

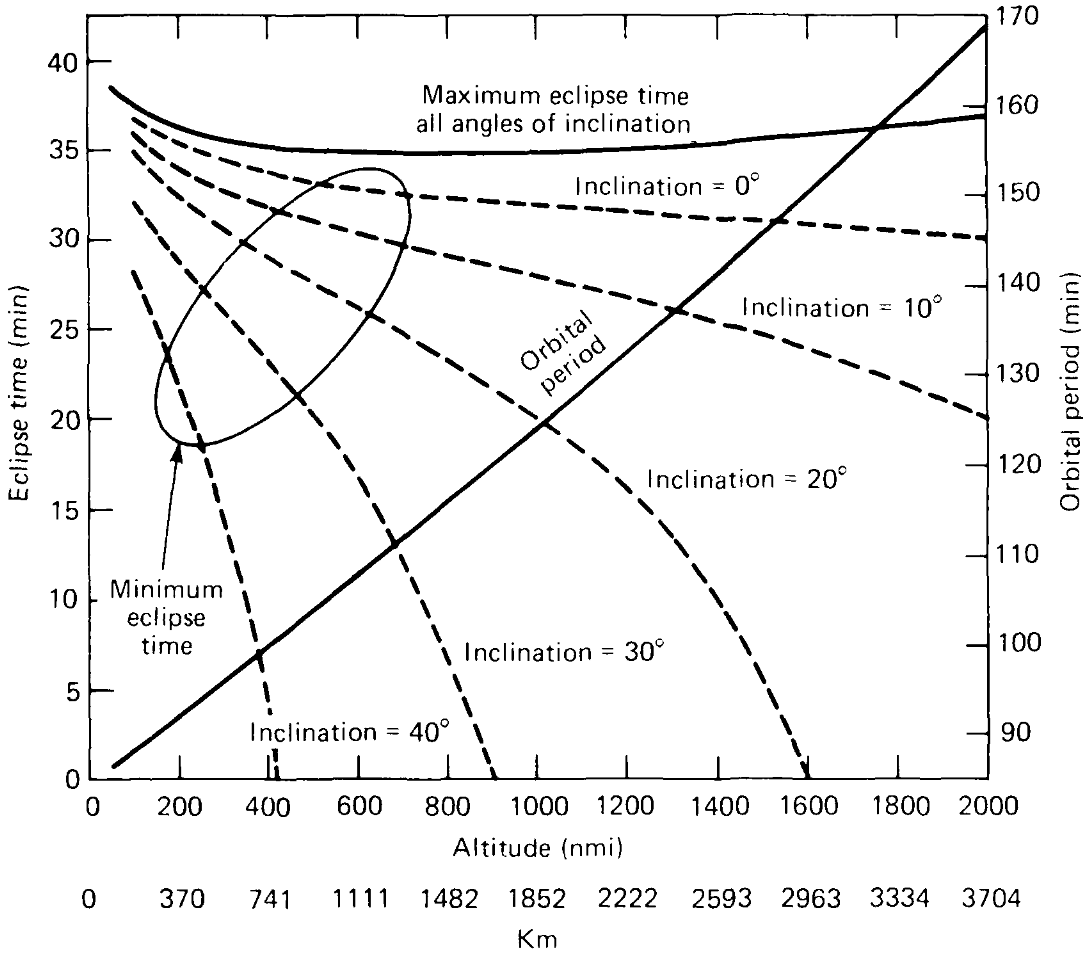


Fig. 6-6. Orbital period, maximum eclipse time and minimum eclipse time for circular orbits.

simplicity this current has been neglected, making the pessimistic case truly worst case while the optimistic case is not completely optimistic.

The maximum, one time, depth of discharge (DOD) calculation is shown in Table 6-5 to have a range of 40 to 74%. This range could be reduced to 32 to 58% by using the transmitter only 50% of the time during the drift/eclipse period. (A similar result can be obtained by using a larger (8 A-H) battery.)

It should be emphasized that this calculation can only be made when the transfer orbit is better defined. Since we can decrease the battery DOD by duty cycling the transmitter, it is unlikely that we will need to go to a larger battery. However, Fig. 6-7 shows available battery sizes and their weights to aid in making this decision at a later time.

6.3.3 Power System Electronics

The SIMPL power system will be required to support an estimated average electrical load of approximately 122 watts during its six year lifetime (see Appendix C). The power subsystem to support this load is shown in Fig. 6-8. It consists of a solar cell array as the basic energy converter, a battery to power the loads whenever they exceed the capability of the array, and a battery charge control system to provide proper battery recharge and to dispose of any excess solar cell energy in such a way as to minimize the impact on the thermal system. The system also has a low voltage sensing switch to prevent excessive discharge of the battery.

Table 6-5

SIMPL maximum battery depth of discharge

Transfer Orbit Load:

Command System	7.8W
Command Receivers	9.0W
Telemetry System	4.7W
Dual Oscillator	1.5W
Time Management Unit	1.0W
Transmitters	27.8W
Power System	5W
Hydrazine Heaters	10W (Est.)
	66.8W (2.4A @ 28V)

Battery Current Drain (90% efficient regulator from 23V)	3.25A

Discharge Time:

	Optimistic Case	Pessimistic Case
Drift Period	45 Min.	45 Min.
Eclipse Period Occurring After Drift Period	0 Min.	37 Min.
Total	45 Min.	82 Min.
Discharge:	2.44 A-H	4.44 A-H

Depth of Discharge:

For 6 A-H Battery	40%	74% (100% Xmtr Duty Cycle)
	32%	58% (50% Xmtr Duty Cycle)
For 8 A-H Battery	31%	56% (100% Xmtr Duty Cycle)

18 cell battery (23.0 V)

Cell size nom. cap A – H	Cell wt (gms)	Battery size				Weight		Rated energy (W – H)	Energy density $\frac{WH}{lb}$
		L (in.)	W (in.)	H (in.)	Volume (in^3/cm^3)	(Kg)	(lbs)		
4	190	10.47	5.024	3.290	173/2836	4.45	** 9.80	92	9.38
* 5	205	10.37	5.006	3.290	171/2799	4.80	10.58	115	10.9
* 6 Tfe	265	11.01	5.012	3.687	203/3334	6.20	13.67	138	10.1
7	310	10.48	5.014	4.460	234/3840	7.25	16.00	161	10.1
8	380	11.02	6.726	3.906	290/4744	8.89	19.60	184	9.38
* 12 Tfe	425?	11.03	6.694	4.151	306/5022	9.95	21.93	276	12.6
* 15 Tfe	635	10.97	6.716	5.392	397/6510	14.86	32.76	345	10.5
* 22 Tfe	820	11.58	8.076	5.470	512/8383	19.19	42.30	506	12.0

*Low profile cells

**Battery = 1.3 × cell wt + baseplate wt
Baseplate for one battery = 207 G = 0.457 lbs (not included in wt. calc.)

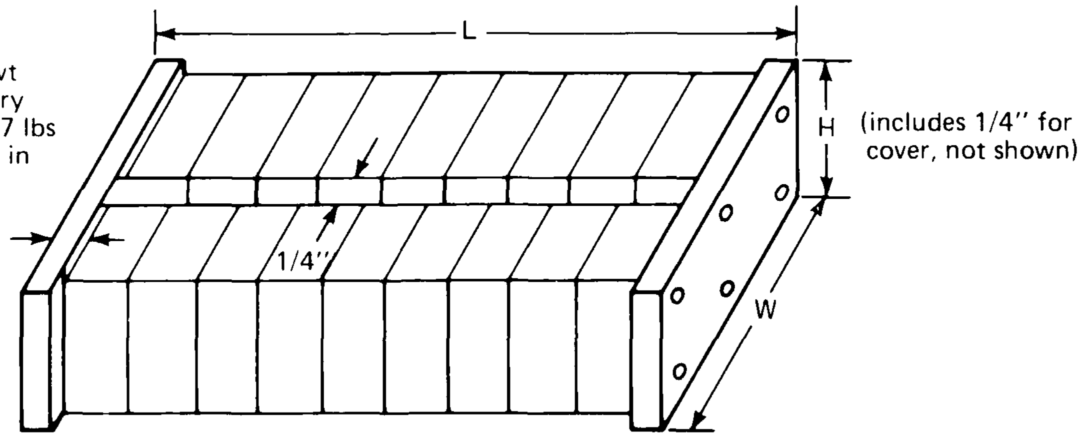


Fig. 6-7 Nickel cadmium battery sizes and weights, based on G.E. cells.

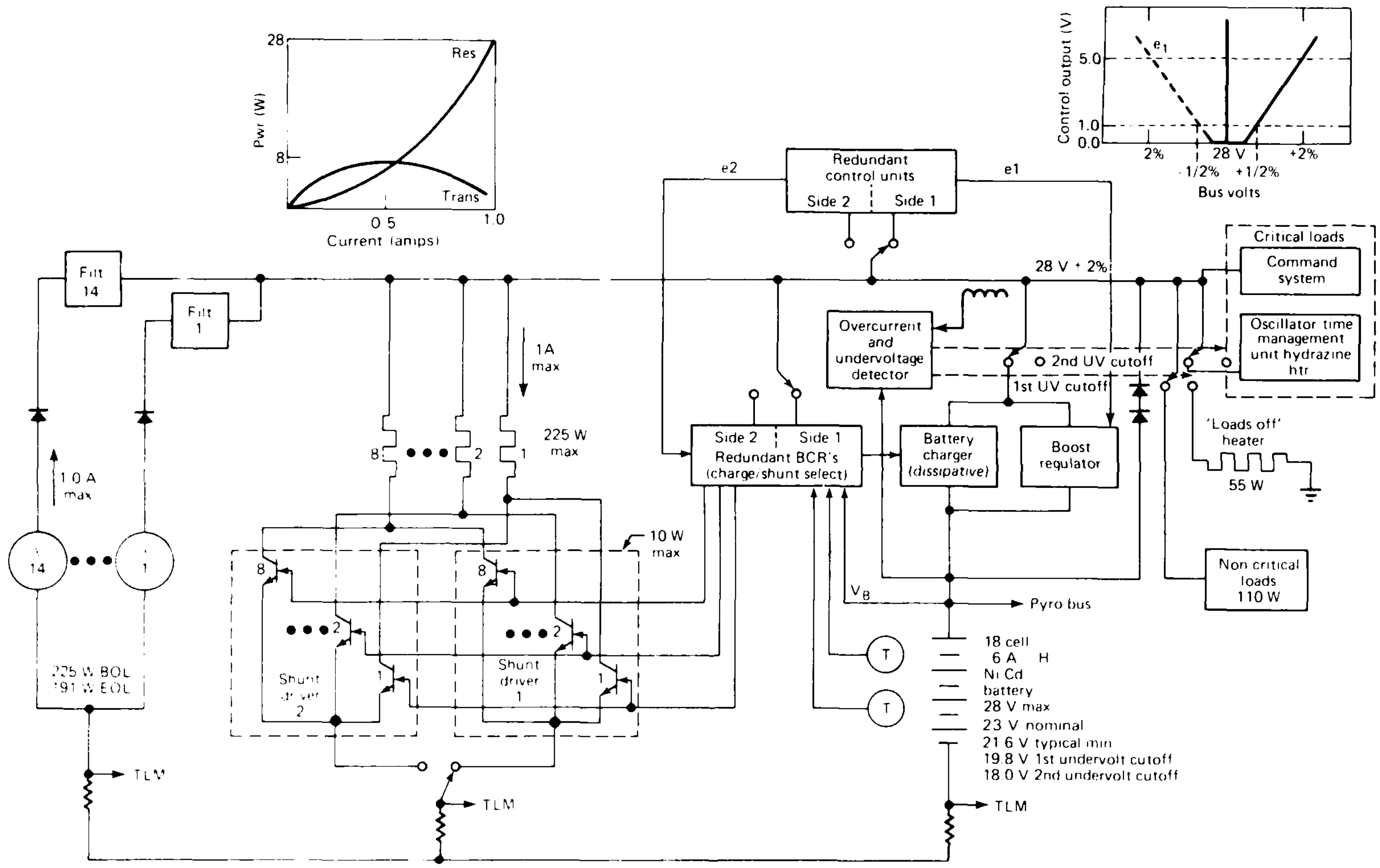


Fig. 6-8 SIMPL DET power system with 'solar only' capability.

6.3.3.1 Heritage

The system is basically a 28V ± 2% DET (Direct Energy Transfer) system developed by GSFC and recently used on the IUE and ISEE-3 satellites. The solar array is very similar to that used on AMPTE. The shunt system is of a type developed by JHU/APL for the SAS-C, MAGSAT, AMPTE, and GEOSAT programs.

6.3.3.2 Battery Charge Control

The heart of the battery charge control system is the controller, which continuously monitors the bus voltage. Whenever the bus voltage drops below 28 volts, a voltage will be developed at e_1. When this voltage exceeds one volt, it will turn on the boost regulator, which will supply sufficient current from the battery to the bus to meet the load demand. When the spacecraft enters the Sunlight the solar array will supply sufficient energy to the bus to cause it to increase above 28 volts. This produces an output voltage e_2 from the controller. When e_2 exceeds one volt, the charge/shunt select will activate the battery charger until the battery voltage limit is reached; then its current will be reduced to a trickle charge. Any excess power generated by the solar array that is not needed for the load or the battery is dissipated in the shunts, which will be designed to have low-power dissipation. There will be four different battery V-T limit curves, selectable by ground command.

6.3.3.3 Overcurrent and Undervoltage Sensing System

An overcurrent and undervoltage sensing switch (OUSS) will be incorporated in the power system to protect the battery from excessive and potentially damaging discharge. In the event

of a low voltage or overcurrent on the main power bus, the OUSS will serve as a circuit breaker and disconnect all non-critical spacecraft electrical loads. In addition to the critical loads (listed below), "loads-off" heaters will be energized to keep the satellite temperature at a safe level until the ground controllers can act. Recovery from an OUSS trip will be performed manually, by ground controllers, so that failures, if they have occurred, may be identified and isolated. The OUSS function can be disabled by ground command if necessary. When enabled, it may be set by command to either of two voltage levels corresponding to approximately 1.10 and 1.00 volts per battery cell.

The critical loads for this mission are:

1.	Command System	7.8W
2.	Command Receivers	9.0W
3.	Dual Oscillator	1.5W
4.	Time Management Unit	1.0W
5.	Hydrazine System Heater	15W (Est.)
6.	Boost Regulator	3W
7.	Battery	4W
	Total	41.3W

The first OUSS trip would cause all loads except those above to be removed. The second trip would cause items 3 through 6 to be removed and the battery to be OR'ed to the bus.

It should be noted that if this OUSS trips, the spacecraft will operate indefinitely in the Solar Only mode as long as the active sides of the solar panels are facing the Sun.

If for some reason the spacecraft were to turn over so that the solar panels no longer faced the Sun, then it would take about one hour for the 122 watt load to cause the OUSS to reach the first trip point. This event is unlikely, but if it is a concern we could put sufficient solar cells on the back of the solar panels and the spacecraft body to supply power to the command system, the receivers, and the thrusters. Another strategy would be to sense reduced array output immediately, rather than wait for the loads to drain the battery. This problem is fundamentally no different from the situation for ISEE-3.

6.4 ATTITUDE DETERMINATION AND CONTROL

6.4.1 Attitude Determination

SIMPL requires that the attitude of the the boom-mounted magnetometer be known to within 1°. Attitude and delta-v maneuvers require comparable attitude knowledge in order to minimize expenditure of propellant. Allowing for sensor misalignments, magnetometer boom misalignments, and attitude solution uncertainties requires that attitude determination be accurate to <0.5°. Consequently, the attitude sensor components must have sufficient accuracy to support solving for the spacecraft attitude to within this accuracy.

6.4.1.1 Attitude Sensors

The selected attitude sensor complement for the SIMPL mission consists of a redundant spinning Sun sensor system and a pair of star scanners. The Sun sensor system will determine the angle between the Sun-spacecraft line and the spacecraft spin axis. The use of star scanners, mounted with their optical axes nearly perpendicular to the spin axis, will enable near real-time attitude determination in both the transfer and halo orbit phases of the mission. Descriptions of the proposed sensors and the reasons for selecting them are given below.

Generally, for an Earth-orbiting spacecraft, a variety of sensor systems may be used for attitude determination. Typically, these might include some combination of Sun sensors, Earth-horizon scanners, star sensors, or magnetometers. SIMPL's Earth-Sun L_1 libration point orbit limits the usefulness of some of these systems. For example, the magnetic field near L_1 is both variable and unpredictable, precluding the use of magnetometers.

ISEE-3, the first Earth-Sun L_1 libration point satellite, utilized a redundant Fine Sun Sensor (FSS) system and a Panoramic Attitude Sensor (PAS) for attitude determination. The PAS is an optical sensor capable of detecting visible light from the Earth, Sun, and for certain geometries, the Moon. The ISEE-3 FSS had an accuracy of 0.1°, while the PAS was accurate to 0.7°. These systems were used together only during the early stages of the transfer orbit, however.

Although the SIMPL orbit is similar to ISEE-3's, SIMPL's attitude during the mission will be very different. After separation from its Delta booster, ISEE-3 required a large angle (approximately 90°) reorientation maneuver to achieve nominal

mission attitude with its spin axis perpendicular to the ecliptic plane. For power reasons it was necessary to perform this maneuver as soon as possible after booster separation at a time when the Earth, Sun, and spacecraft were very nearly colinear. Consequently, the Sun and Earth could not both be used as attitude references and another reference was necessary. This led to the choice of the PAS to be used for Moon detection during that crucial attitude maneuver since, at the time, star scanners were significantly more expensive. Dependence upon the Moon as an attitude reference necessarily constrained the launch date to a few days each month. This was not an unbearable constraint for ISEE-3's dedicated Delta launch. But for the Shuttle launch baselined for SIMPL, where the manifest will be shared with several other spacecraft, it is preferable to minimize additional launch constraints.

In the L_1 halo orbit, the Earth-spacecraft and Sun-spacecraft lines are very nearly parallel, resulting in poor geometry for attitude determination. Ideally, the two vectors used for attitude determination should be perpendicular to one another in order to minimize uncertainties in the solution process. The ISEE-3 approach was to use FSS data integrated over a 14-28 day period to solve for the attitude during the data collection period. Consequently, the most recent attitude solutions were always 14-28 days old. The PAS, used only during the ISEE-3 transfer orbit, was essentially useless at L_1, yet its mass had to be carried throughout the mission.

While the ISEE-3 attitude determination approach worked, it is not a particularly attractive method since one device, the PAS, was useless in the mission orbit, forcing attitude estimates to be obtained from FSS data alone, 14-28 days after the fact. In addition, the PAS was originally designed in the early 1970's

without the radiation hardening and redundancy required for this mission. Furthermore, its 0.7° resolution is inadequate for SIMPL. Finally, the PAS is no longer produced by Ball Brothers Research Corporation, except as a special order item, making its price now comparable to a much more accurate star scanner. The choice to be made for SIMPL attitude determination, then, is whether to augment Sun sensor data with Earth scanner data during the early phases of the transfer orbit and accept 14-28 day delays for attitude determination in the halo orbit, or whether to carry a star scanner system which can be used during the entire mission to provide attitude updates. Earth scanner data is very simple to process but, as with the PAS, would be essentially useless at L_1 distances. Star scanners, on the other hand, are accurate virtually anywhere during the SIMPL mission but typically require a great deal of processing to extract star sightings. Generally, an initial estimate of the spacecraft attitude is necessary to narrow down the list of stars that might be identified by a star scanner.

Because SIMPL's mission dictates that attitude information be updated more frequently than once every two weeks under the unfavorable L_1 geometry conditions, we concluded that a star scanner would be needed to supplement Sun sensor data for attitude determination. Because of the crucial importance of attitude data, we have included a pair of star scanners to provide redundancy and ensure that one will be operational for the entire mission. The spare star scanner will be unpowered when not required.

A particular star scanner has not yet been identified for the SIMPL mission. However, during JHU/APL's study for NASA/GSFC of the ISTP/WIND spacecraft, Ball Brothers indicated that one of their designs, a version of their model CS-203 solid state star

scanner, might be appropriate. The CS-203 variant considered for WIND would have required modification to operate at WIND's 20 rpm spin rate, having been designed for a nominal spin rate of six rpm. The proposed spin rate for SIMPL, however, means that little or no modification would be required for inclusion of the CS-203 on SIMPL. The CS-203 or its variant, with an accuracy of 0.1°, will be mounted with its line of sight nearly perpendicular to the spin axis, canted slightly to keep the booms out of the field of view.

The Sun sensor system selected for SIMPL is the redundant Digital Solar Attitude Detector (DSAD) system used on our AMPTE spacecraft. It has a Sun elevation angle accuracy of 0.5° in the range from 5-175°. A Sun crossing pulse is generated as the Sun passes through the sensor's field of view. The DSAD system consists of two sensor heads and a fully redundant set of electronics. The sensor heads will be mounted on the spacecraft such that their planar fields of view look radially outward from the spacecraft, with the plane containing each field of view also containing the spacecraft X-axis.

6.4.1.2 Attitude Processing

The star scanner and DSAD data will be used to solve for SIMPL's attitude. Because of the low bit rates available for transmitting instrument, attitude, and housekeeping data to Earth, it will be necessary to do some processing of the attitude sensor data onboard the spacecraft. In addition, the Sun sensor information will be required onboard to determine the proper Sun-relative azimuth phasing of the precessional thruster firings. If possible, it is desirable to do all of the attitude processing onboard SIMPL so that the final attitude estimate can be incorporated directly in the downlink data product.

The DSAD output consists of a Sun crossing signal and a Gray-coded Sun elevation angle. The Sun crossing signal will be used to generate a Sun crossing pulse sent to an Azimuth Signal Generator that will generate 4096 equally spaced sector pulses each revolution of the spacecraft. The sector pulses will be used to phase firings of the thrusters so that ground-commanded precessional maneuvers of the spin axis can be effected. The Gray-coded elevation angle and the corresponding Sun crossing time will be used in the attitude determination algorithm.

As mentioned in the previous section, processing of star scanner data is nontrivial. The star scanner contains two slits, nominally a V-shaped pair for the CS-203, for detecting stars. Each star scanner ideally generates two pulses each time its field of view sweeps past a star that is brighter than a preselected threshold. The sensor outputs transit times for each pulse and the measured intensity of the star. The star can then be uniquely identified by its magnitude, and its azimuth and elevation relative to the spacecraft can be determined from the two crossing times.

Unfortunately, star sightings and identification are not ideal. The star scanner output must be matched with a star in a star catalog; the larger the catalog the more difficult it becomes to uniquely identify the sighted star. Also, the detected star intensities generally do not correspond exactly to the magnitudes of any star in the catalog. To further complicate matters, false sightings frequently occur from glint, the electronics, or other sources. Consequently, before an attitude determination algorithm can be implemented, each star observation must be checked against all possible catalog stars to resolve, unambiguously, if the sighted star matches a star in the catalog.

To expedite attitude determination, on-board pre-processing will be used to identify detected stars from the star scanner observations. An on-board star catalog will be updated as the background star field changes due to commanded precessional maneuvers of the spin axis. If the computational burden is not prohibitive, this same processor may also be used to completely determine the attitude in addition to identifying stars. Investigation of the processing requirements to completely solve for the attitude from star sightings and DSAD data is beyond the scope of this study and must be left as a task for the next phase of the program.

6.4.2 Attitude Control

The SIMPL attitude control system (ACS) must maintain the spin axis to within 1° of a desired orientation. Open-loop precessional control will be achieved by ground-commanded thruster firings. Active closed-loop nutational control will be required following release from the Shuttle prior to PAM-D separation and appendage deployment. In the mission orbital configuration with appendages deployed, the spacecraft must be passively spin stabilized. The SIMPL ACS that satisfies these objectives is described below.

Upon release from the Shuttle prior to PAM-D separation and subsequent boom deployment, SIMPL will be in an unfavorable moment of inertia configuration, spinning about its axis of minimum moment of inertia. Consequently, on-board active nutation damping will be required to ensure attitude stability. The active nutation damping system is composed of a pair of accelerometers, mounted near the periphery of the spacecraft, and redundant nutation damping electronics. The nutational motion sensed by the accelerometers will be damped by appropriate firings of the

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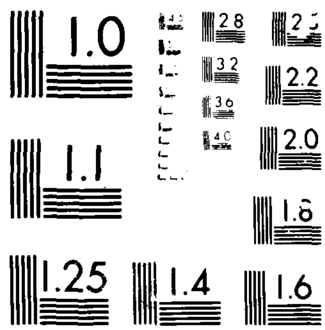
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1.8
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1.4
1.6

attitude control thrusters as commanded by the closed loop damper logic. The exact form of the control logic will be the subject of future study, but should be relatively straightforward to implement. Once a stable, spinning spacecraft configuration is achieved, the active nutation damper will be disabled.

Shortly after PAM-D firing and subsequent separation, the solar panels, magnetometer and inertia booms, and the radial wire antennas will be deployed to their mission positions. In the mission mode, SIMPL must be a passively spin-stabilized, nonrigid spacecraft spinning about its axis of maximum moment of inertia. A favorable ratio must be maintained between the spin and transverse axes' moments of inertia, considering both fuel migration and failure of the radial wires to deploy. Deflection of the deployed wire antennas should not affect attitude stability.

It is not possible, within the scope of this configuration study, to specify the SIMPL spin rate and moment of inertia ratios required to ensure that the spin axis will stay within 1° of the commanded pointing direction during the 11-30 day drift arcs. Preliminary calculations, however, indicate that a five rpm spin rate may be adequate for SIMPL. The external torques, which are to a large part configuration dependent, will be the predominant influence on pointing performance, but these cannot be specified at this time. Solar pressure, the major external disturbance contributor, must necessarily remain an item of future investigation. Natural resonances that might adversely couple with the attitude motion, degrading pointing accuracy, must be avoided but cannot be known without further analysis.

Nutation damping in the mission configuration will be achieved through use of passive nutation dampers. The ball-in-tube nutation dampers used on our AMPTE spacecraft are currently being baselined for SIMPL. Their suitability for SIMPL will be examined in the next study phase. Partially filled fluid dampers will also be evaluated for any potential advantages they may offer to the SIMPL mission.

Pointing of the SIMPL spin axis will be achieved in an open-loop fashion by stored or ground commanded firing of the attitude control thrusters. Desired precessional torques will be generated by firing the thrusters at particular phases in the spin cycle. SIMPL's spin axis will have to be precessed an average of about 1°/day in order for it to remain nominally pointing towards the Earth. Operationally this will be accomplished by ground commanded precessional maneuvers once every 11-30 days as described in Section 4.

6.5 PROPULSION SYSTEM

Liquid propulsion will be used to insert SIMPL into its operational halo orbit about the L_1 libration point. Elements of the same propulsion system, used in various combinations, will also provide spin-axis precession (attitude control), spin rate control, and orbit-adjustment capabilities. The total delta-v (velocity increment) to satisfy all these objectives for six years was shown in Section 4.6 to be 610 m/s. The maximum delta-v capability for the SIMPL propulsion system described below is 791 m/s. To help achieve the six-year lifetime goal, a fully redundant system is provided. Table 6-6 summarizes the design parameters of this system.

Table 6-6

SIMPL attitude, spin, and velocity control propulsion

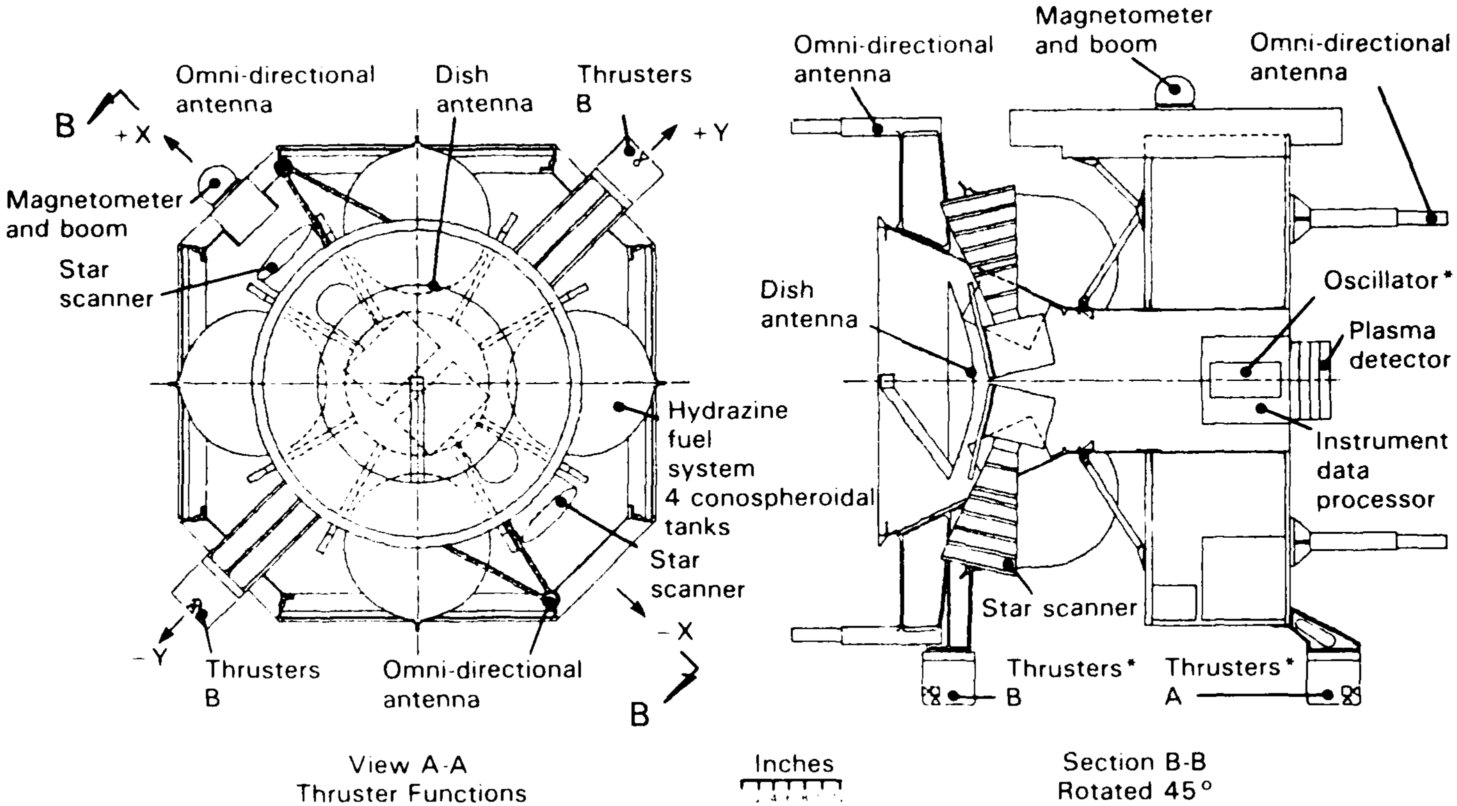
Total Delta-v Required	610 m/s
Spacecraft Wt. (fueled)	393 kg
Propellant	Hydrazine (N_2H_4)
Propellant I_{sp} (eff.)	215 lbf-sec/lbm
Propellant Required	98.7 kg
No. of Tanks	4
Total Tank Capacity	123 kg
Delta-v Capability	791 m/s
Delta-v Margin	+30%
Blowdown Ratio	4:1
System Dry Wt.	16.8 kg
System Fueled Wt.	141.5 kg
Thruster Range	7.5 to 2.0 lbf
Thrusters Required	12

6.5.1 Propulsion System Details

The fuel, monopropellant hydrazine (N_2H_4) with an effective specific impulse I_{sp} of 215 lbf-sec/lbm, is stored under pressure in four conospheriodal tanks symmetrically positioned around the spacecraft Z-axis and located beneath its lower deck (see Fig. 6-9). These tanks (Fig. 6-10), originally developed for the GOES spacecraft, have been flown on the Shuttle-launched SBS-C and SBS-D, and are included in the designs of numerous, yet-to-be-launched Shuttle spacecraft. Use of an already Shuttle-qualified tank saves substantial costs.

Designed specifically for use on spinning spacecraft, the tanks rely on centrifugal force, rather than elastomeric diaphragms, to keep the pressurant (gaseous nitrogen, GN_2) from mixing with the propellant when the system is operational. Because it is less dense than N_2H_4, the GN_2 migrates toward the spacecraft spin axis and away from the thruster feedline port located at the outward-pointing apex of the tank.

Thruster and safety valve actuation causes N_2H_4 to flow under pressure into the thruster catalyst bed, where it decomposes into ammonia, nitrogen, and hydrogen. This process produces sizeable increases in the temperature and pressure of the byproducts as they enter and pass through the nozzle to produce thrust. The "blowdown method" is used to feed the N_2H_4 into the thruster feedlines, meaning that the pressure behind the N_2H_4 decays from some high initial level to lower levels as fuel is consumed. Since thrust level is directly related to feedline pressure, thrust decays with system use. Consequently, thrust at the end of the mission will be considerably less than it will be at the start of the mission. Since impulse rather than thrust



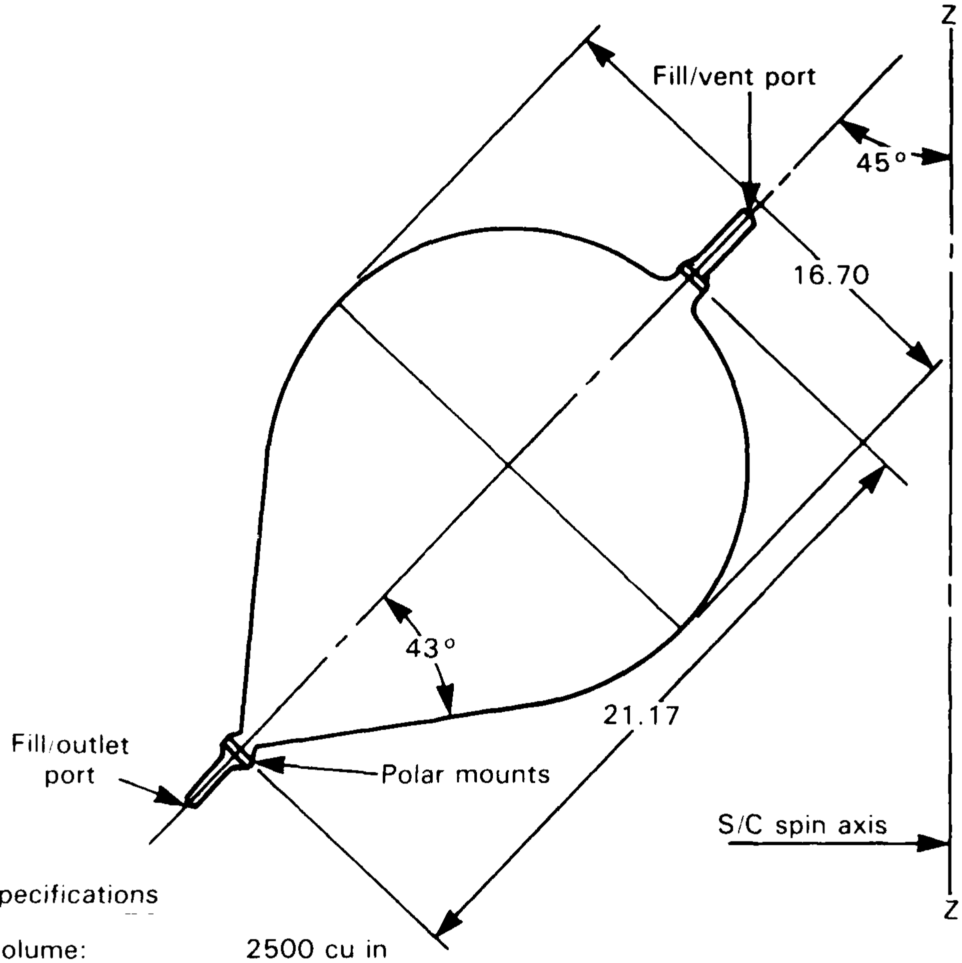
(2) A {1 Velocity control
1 Attitude control/velocity control

(2) B {1 Velocity control
1 Attitude control/velocity control
2 Spin control

Total no. of thrusters = 12

*Components rotated into view

Fig. 6-9. SIMPL thruster and propellant tank arrangements.



Tank specifications

Total volume:	2500 cu in
Propellant volume:	1886 cu in
Operating pressure:	350 psi
Proof pressure:	567 psi
Burst pressure:	700 psi
N_2H_4 capacity:	68 lbs
Wall thickness:	.020
Material:	Ti 6Al 4V
Weight:	4.10 lb
No. tanks required:	4

Manufacturer: Fansteel Precision Sheet Metal
Fansteel Dwg No. E4425078A

Note: 1. This tank is Shuttle qualified. Shuttle fracture mechanics criteria have been met.
2. Shuttle flight history: SBS-C 11/11/82. SBS-D,E,F to be Shuttle launched in near future.

Fig. 6-10. SIMPL conospheroidal propellant tank.

establishes the end result, the decay in thrust level throughout the mission is compensated for by periodically increasing thruster burn time.

To minimize destabilizing torques during thruster operation, spacecraft center-of-mass drift must be confined to a relatively narrow range. The principal contributor to center-of-mass drift is the reduction of fuel mass as fuel is consumed. Positioning the fuel tanks symmetrically about the spacecraft Z-axis eliminates the potential for intolerable center-of-mass shifting normal to that axis so long as fuel is depleted fairly symmetrically from opposite pairs of propellant tanks. This should not be a problem for SIMPL. For those thrusters that are sector fired in pairs pointing radially outward from the spacecraft and in the same direction, center-of-mass drift is more important. Unless the tank system can be located such that its fuel mass center remains near the spacecraft center-of-mass throughout the mission, pairs of thrusters firing simultaneously and in the same direction will create unwanted attitude disturbing torques. These disturbances can be controlled by having each thruster deliver a slightly different impulse, i.e., one of the thrusters must fire over a longer duration.

6.5.1.1 Propulsion System Elements

The SIMPL propulsion system is comprised of the components tabulated in Table 6-7. Figure 6-11 shows these components functionally identified in schematic form. For clarity, the tank heaters, filters, and manifolding are not shown.

Table 6-7

SIMPL propulsion system components

Item	Quantity	Remarks
Propellant Tanks	4	N_2H_4 capacity 30.7 kg pressurized to 350 psi; two sets of two tanks cross-coupled for leak isolation and improved reliability.
Thrusters	12	Blowdown from 7.5 to 2.0 lbf.
Latching Valves	12	Draw no power once opened.
Thruster Valves (Fail Safe)	12	Satisfy STS power failure safety requirement.
Isolation Valves (Latching)	4	Isolate propellant tanks from thruster feed lines for leak isolation and cross-coupling capability.
Fill-Vent Valves	2	G_2N fill and vent.
Fill-Drain Valves	2	N_2H_4 fill and drain.
Pressure Transducers	2	Monitor pressure in each tank circuit.
Tank Heaters	4	Keep N_2H_4 from dropping below freezing temperature (0° C).
Filters	6	Filter fuel upstream of each isolation valve and main thruster lines.
Plumbing	-	Ti6Al4V with welded connections.
Manifolding	-	Provides mounts for thruster assemblies.

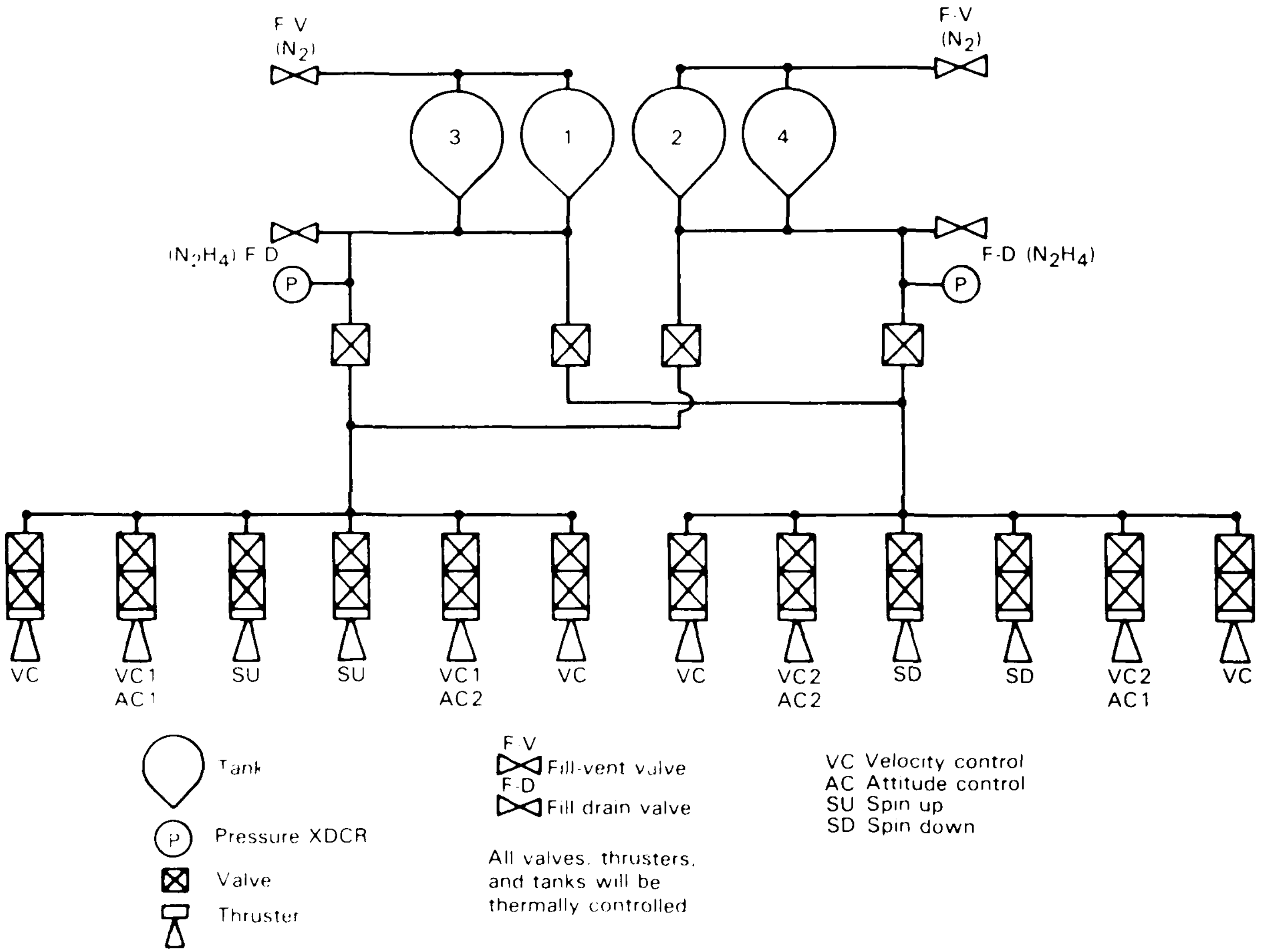


Fig. 6-11. **SIMPL propulsion system schematic.**

6.5.1.2 Propellant Tanks

The conospheroidal propellant tanks are mounted symmetrically around the spacecraft Z-axis. Each tank is canted 45° relative to the Z-axis as shown in Fig. 6-10. The tank shape and orientation are dictated by the necessity to drain all of the tanks with the spacecraft sitting in one position. When the spacecraft spins, centrifugal force moves the much denser N_2H_4 radially outward, forcing any dissolved GN_2 to bubble into the spherical portion of the tank. The N_2H_4 fills the thruster feedlines at an initial pressure of 350 psi. This pressure decays steadily as the fuel is consumed and drops to 86 psi when the fuel is totally depleted, for a blowdown ratio of 4:1. The centrifugal force contributes very little to the pressure head provided by the GN_2 (less than 1 psi at 60 rpm).

Energy dissipation due to motion of the fuel relative to the tank walls will be of beneficial value whenever the spacecraft nutates during and after thruster activity, because the spacecraft is dynamically stable relative to its axis of rotation.

The tanks are provided with polar mounts which facilitate their installation in the spacecraft. The requirement that they be mounted in the spacecraft as depicted in Fig. 6-10 to facilitate N_2H_4 draining introduces a complication when the spacecraft is mounted in the Shuttle cradle. In this instance, since the spacecraft Z-axis is perpendicular to the Shuttle longitudinal axis, draining of N_2H_4 in the Shuttle cargo bay is impossible whenever the Shuttle is in the gantry. In an emergency the spacecraft would have to be removed from the cargo bay and set upright before fuel removal could begin. Fuel draining is never attempted when the system is under pressure. Removing GN_2 saturated with hydrazine vapor creates a containment and disposal

problem which must be addressed with special ground support equipment. Venting contaminated GN_2 to the atmosphere in the Shuttle cargo bay or anywhere else at the launch site will be forbidden.

[illegible] operational pressure and very close to the 5[illegible] psi proof pressure [illegible]

6.5.1.3 Thrusters

[illegible] thrusters of the Hamilton Standard REA 16 or REA 17-12 type or their equivalents will be used in various combinations to provide the spacecraft with velocity, attitude, and spin control. These thrusters are rated at [illegible] and 1 lbf, respectively. The thrust range over which the REA 16 operates in a blowdown mode is 7.5 to 2.0 lbf. REA 17-12 [illegible] range of 1.0 to 0.4 lbf. The blowdown pressure [illegible] psi.

6.5.1.4 Valves

Shuttle safety rules require three valves in series for each thruster, one of which must be fail-safe. The peak power required to open the valve associated with the larger thruster is

13.5 W while the small thruster valve requires only 3.7 W. Under normal operation the thrusters will be fired in pairs, which means [illegible] watts of peak power will be required to fire the 5 lbf thrusters. [illegible] peak power [illegible] battery [illegible]

[illegible]

[illegible] "B" clusters are similar to the "A" clusters but have in addition two [illegible] spin-control thrusters mounted [illegible] their manifolds.

The spin-control thrusters can be fired individually or in opposed pairs. To achieve the halo orbit, one pair of velocity

control thrusters in either the "A" or "B" cluster will be fired. Attitude maneuvers will require firing an opposite pair of "A" and "B" radial thrusters over predetermined segments of revolutions of the spacecraft. Velocity control will also be possible by firing in a similar manner a pair of "A" and "B" radial thrusters on the same side of the spacecraft. This "sector-firing" is controlled by the Sun-pulsed digital solar attitude detectors and the attitude control processor. Velocity control achieved through sector-firing will be employed after the spacecraft has been placed into its Sun-facing attitude.

6.5.2 Shuttle Safety Policy and Requirements

The SIMPL propulsion system must comply with the requirements stated in Refs. 6-1, 6-2, and 6-3. Reference 6-3 is used as a source of design guidelines rather than a requirements document.

Liquid propellant propulsion systems are classified as catastrophic hazards and, therefore, no combination of two failures, operator errors, or RF signals can result in the potential for personal injury or loss of the Orbiter, ground facilities, or STS equipment.

Each propellant feedline must contain a minimum of three mechanically independent propellant flow-control devices in series. These devices must remain closed during all ground and flight phases until the payload has reached a safe distance from the Orbiter, following deployment. A flow-control device must isolate the propellant tanks from the remainder of the distribution system. One of the three flow-control devices must be fail-safe, i.e., loss of power must interrupt flow.

The propellant tanks satisfy the provisions of Ref. 6-2 and have an ultimate safety factor of two. Pressurized lines and fittings will have an ultimate safety factor of four.

6.6 MECHANICAL DESIGN

The spacecraft composite design, including the arrangement of all packages and subsystems, is shown in Figures 6-12 through 6-15. The idea of using the existing AMPTE/CCE structural design as a basis for SIMPL has proven to be eminently successful. Only modest changes were required to accommodate the SIMPL instrument complement, propulsion system fuel tanks and thrusters, and unique electronic packages.

The most visible difference between the SIMPL and AMPTE structures is the addition of a conical adapter section to mate to the PAM-D and provide room for the high-gain antenna and some of the packages. Other changes are easily handled by the AMPTE design with maximum use of AMPTE hardware. For example, the slightly larger solar cell array required by SIMPL is easily obtained by increasing the length of the AMPTE substrates. No changes in array hinges, panel retention system, or despin cable design will be required. This approach of maximizing heritage was applied throughout and will yield significant cost savings.

Packages have been arranged for easy access and minimal length of interconnecting electrical harness. Provision has been made for a low-EMI (extensively shielded) harness to satisfy the Radio Wave instrument. Ample room is provided for securing all electrical harness bundles. Forward and aft equipment platforms are easily disconnected by using the clustered cable feedthrough. The RF packages are grouped near the Earth-facing antennas to minimize cable losses.

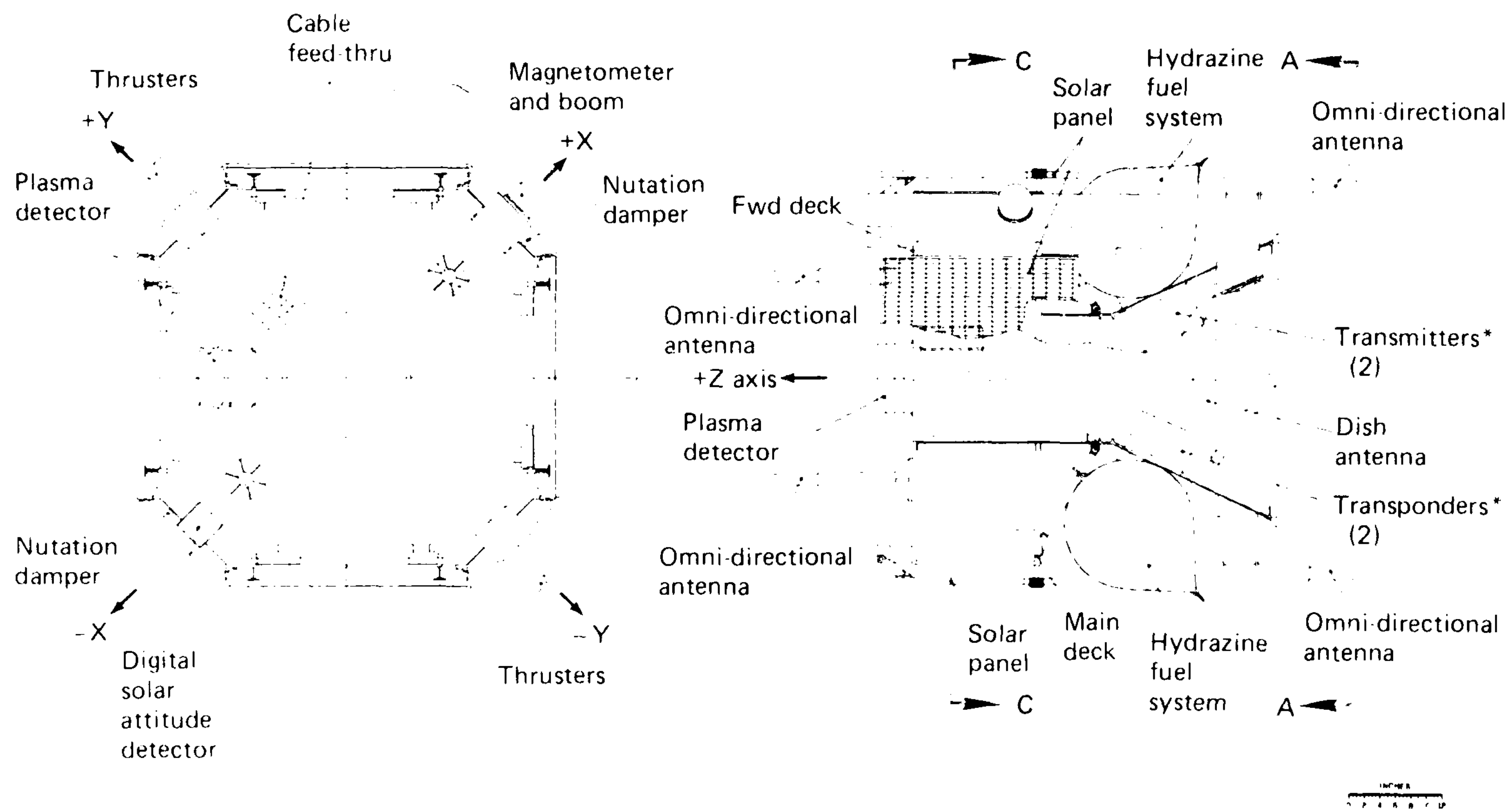


Fig. 6-12. **SIMPL configuration.**

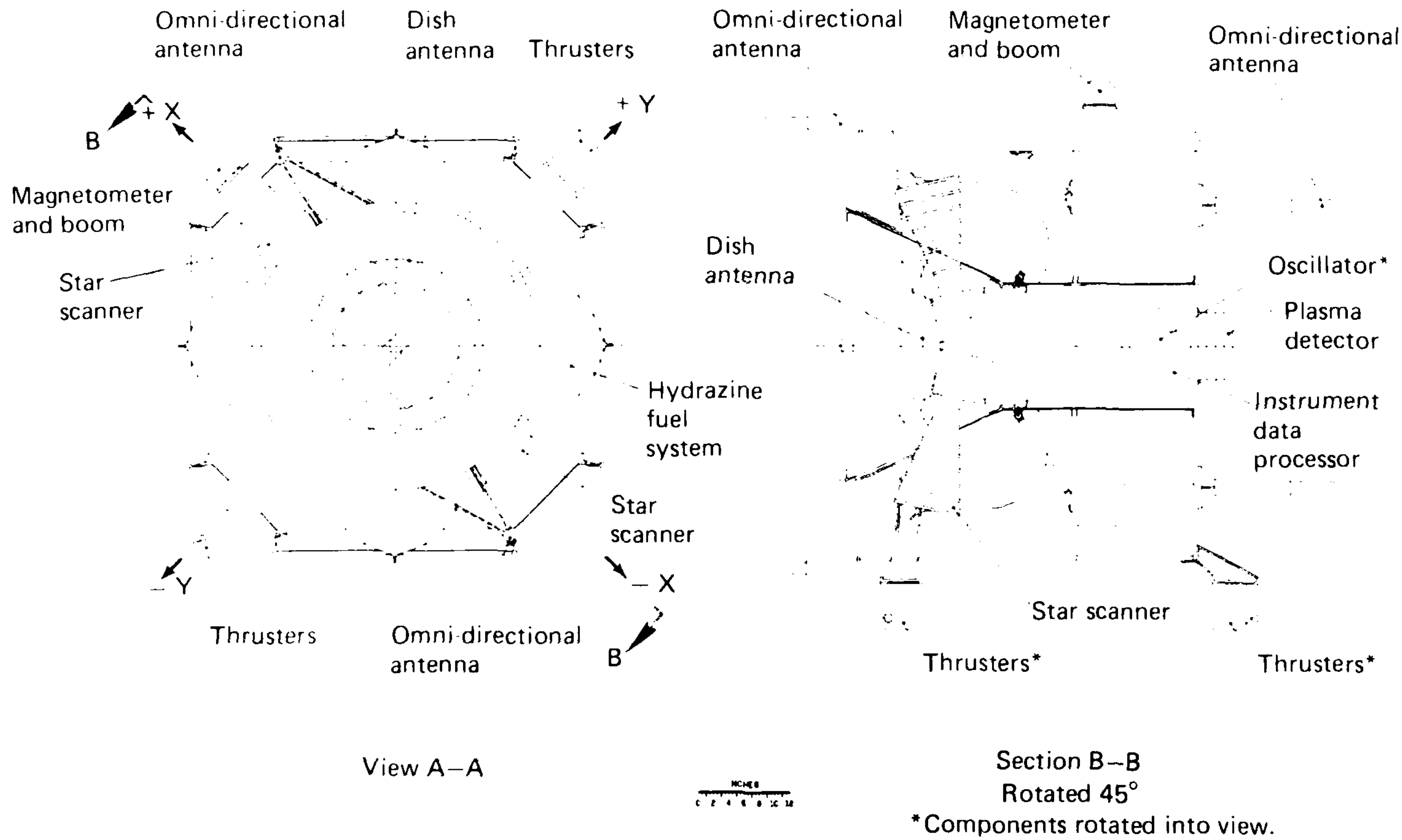


Fig. 6-13. View AA of Fig. 6-12.

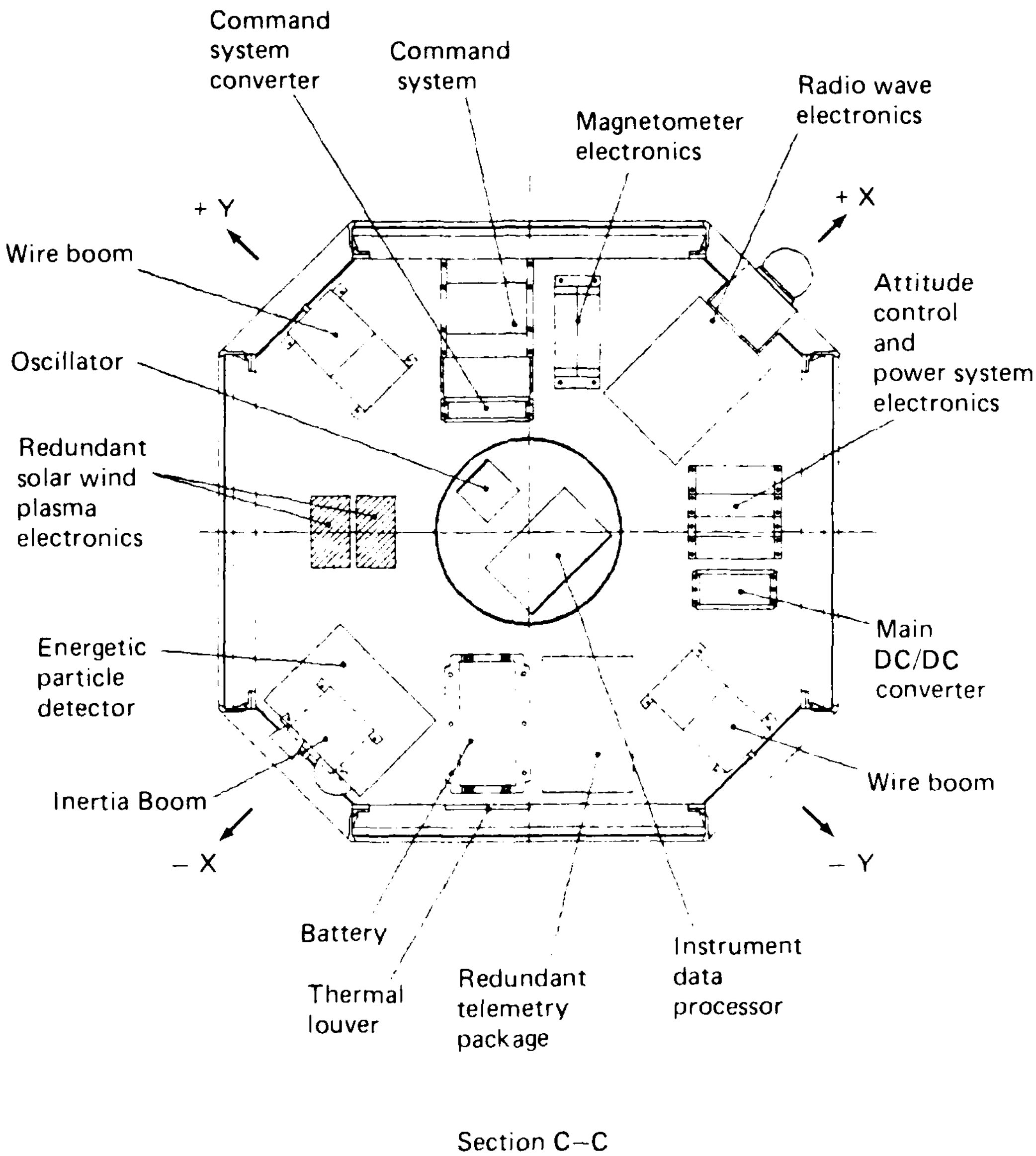


Fig. 6-14. View CC of Fig. 6-12.

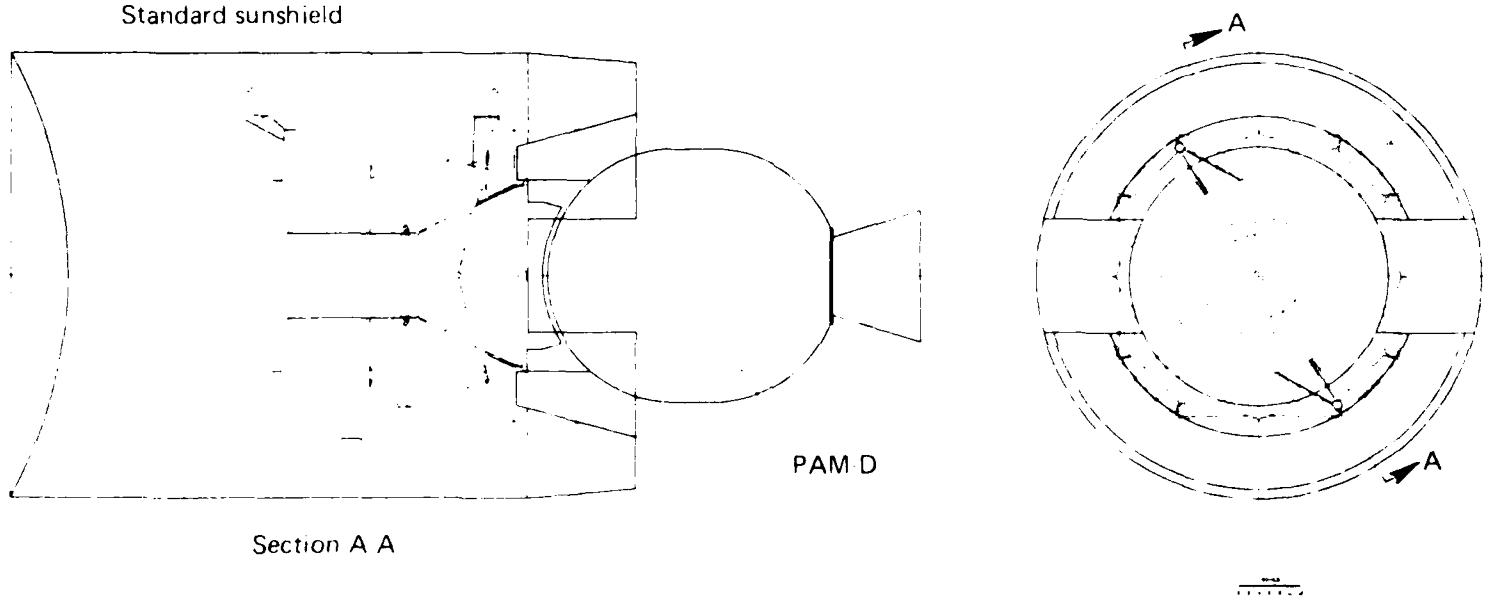


Fig. 6-15.   SIMPL launch configuration.

The battery, with its permeable nickel-containing cells, has been located as far as possible from the magnetometer sensor. A side-facing thermal louver has been provided for, if further analysis shows it is needed. The main DC/DC [illegible] has been located between the redundant telemetry packages and the attitude control and power system electronics [illegible] motor and cold gas systems [illegible] AMPI [illegible] for packages. This choice location, [illegible] radiation and thermally stable, was [illegible] processor and the oscillator.

The four hydrazine fuel tanks, [illegible] transmitters, and [illegible] transponders make use of the large [illegible] structure [illegible] Earth-facing end) as a thermal radiator. Thermal [illegible] is provided between the high-powered RF equipment and the tanks. The tanks further utilize the solar array substrates as [illegible] to minimize the amount of multilayer thermal insulation required. Note that MLI and the propulsion plumbing runs have been deleted from all configuration drawings and the artist's concept (frontispiece) for reasons of clarity.

Center-of-gravity location and moments of inertia can easily be trimmed within the overall spacecraft weight limitation. Specific weight is carried for this purpose.

6.6.1 Deployed Appendages

Items requiring deployment have been kept to an absolute minimum, i.e., the magnetometer boom, inertia boom, and two wire booms. No portion of any of the five antennas requires deployment. Vee-band separation and despin weight cable initiation feature proven electroexplosive devices used on many previous launches.

Two options were available to achieve the necessary separation of the magnetometer sensor from the spacecraft main body. The original AMPTE magnetometer boom consisted of two 4 ft links yielding an 8 ft separation. Extending that boom concept to four links (16 ft) was considered to be about the maximum practical limit due to launch stowage and boom rigidity requirements. Since 20 ft separation was desired, we selected instead a simplified version of our MAGSAT scissors type boom. The gimballed base and complex optical attitude transfer system required on MAGSAT are not needed for SIMPL. The equipment decks will be notched slightly to accommodate the folded boom within the AMPTE despin cable design. A tape type inertia boom with small end mass counterbalances the magnetometer boom and helps raise the Z-axis moment of inertia.

The Radio Wave antennas are shown as motor-fed wire booms with small end masses to aid deployment. The same type was used on ISEE-3.

6.6.2 Launch Accommodations

Space Transportation System (Shuttle) safety and planning requirements are stringent, but well known to us. They will be felt in cost and schedule impact in varying degrees of severity and impact the mechanical design as follows:

1. A fracture control plan must be implemented, and fabrication of hardware must define an adequate amount of non-destructive testing, such as dye-penetrant inspection.

2. Aluminum alloys such as 2025 must be avoided, with substitution of 6061 to avoid stress-corrosion cracking concerns. Likewise, 7075 alloys must be over-aged with proper heat treatment.

3. Fuel tanks must be drainable in a one-g static environment.

4. Nonmetallic materials must be carefully selected from approved lists.

The launch configuration (Fig. 6-15) shows SIMPL on an off-loaded (baseline) PAM-D upper stage with no violation of restricted volume below the separation plane. Even with the smallest standard Sunshield (shown), SIMPL does not fully utilize either the volume within the Sunshield nor the loft capability of the PAM-D motor. The opportunity for launching a substantial subsatellite may be of interest to the Air Force.

6.7 THERMAL DESIGN

The thermal design of SIMPL is based on the design of AMPTE/CCE. The AMPTE design used anti-Sun facing radiators with louvers; the rest of the surfaces were covered with multilayer insulation (MLI). In addition, an aluminum shield on the Sun-facing side protected the insulation from the solid propellant inclination adjust motor. On AMPTE the louvers were needed because of the large change in internal power from the initial orbit to the final orbit.

For SIMPL the multiple louvers are not needed. The launch scenario consists of ejection from the Shuttle bay, a short (45 min) coast period, the PAM-D firing, and immediate

reorientation of the spacecraft spin axis toward the Sun. This period is sufficiently short that the temperatures will change little due to the inherent thermal lag of the spacecraft. At the same time, nearly nominal power will be established immediately; thus the need for general control by louvers is avoided. However, significant changes in battery overcharge may occur. Thus there may be a need for a single louver for battery temperature control. Space has been reserved for a side facing louver near the battery, and it will be considered more fully as the detailed design proceeds. To provide a conservative cost estimate, the louver costs have been included.

The thermal design is shown schematically in Fig. 6-16. Sun facing and side surfaces are covered with 20 layers of MLI, the outer layer of which is heavier and coated with conductive black paint. There are four radiators of one layer of Kapton in the side insulation. The area of these radiators is approximately 8 ft^2, the exact size being easily changed to accommodate the actual internal load as necessary. The battery louver (if needed) will be in a side panel.

Because of the length of this mission and the energetic particle environment, there are no stable thermal control surfaces with low solar absorptance. However, this is not a problem because the radiators do not face the Sun directly and the range of Sun angles is small (0-20° after the transfer phase). Further, the radiators are located as much as possible under the solar panels, which then provide shade at the higher Sun angles.

In order to successfully execute a passive design, the internal power dissipation must be constant within about 20%. Because of the continuous operation of the RF transmitters, and

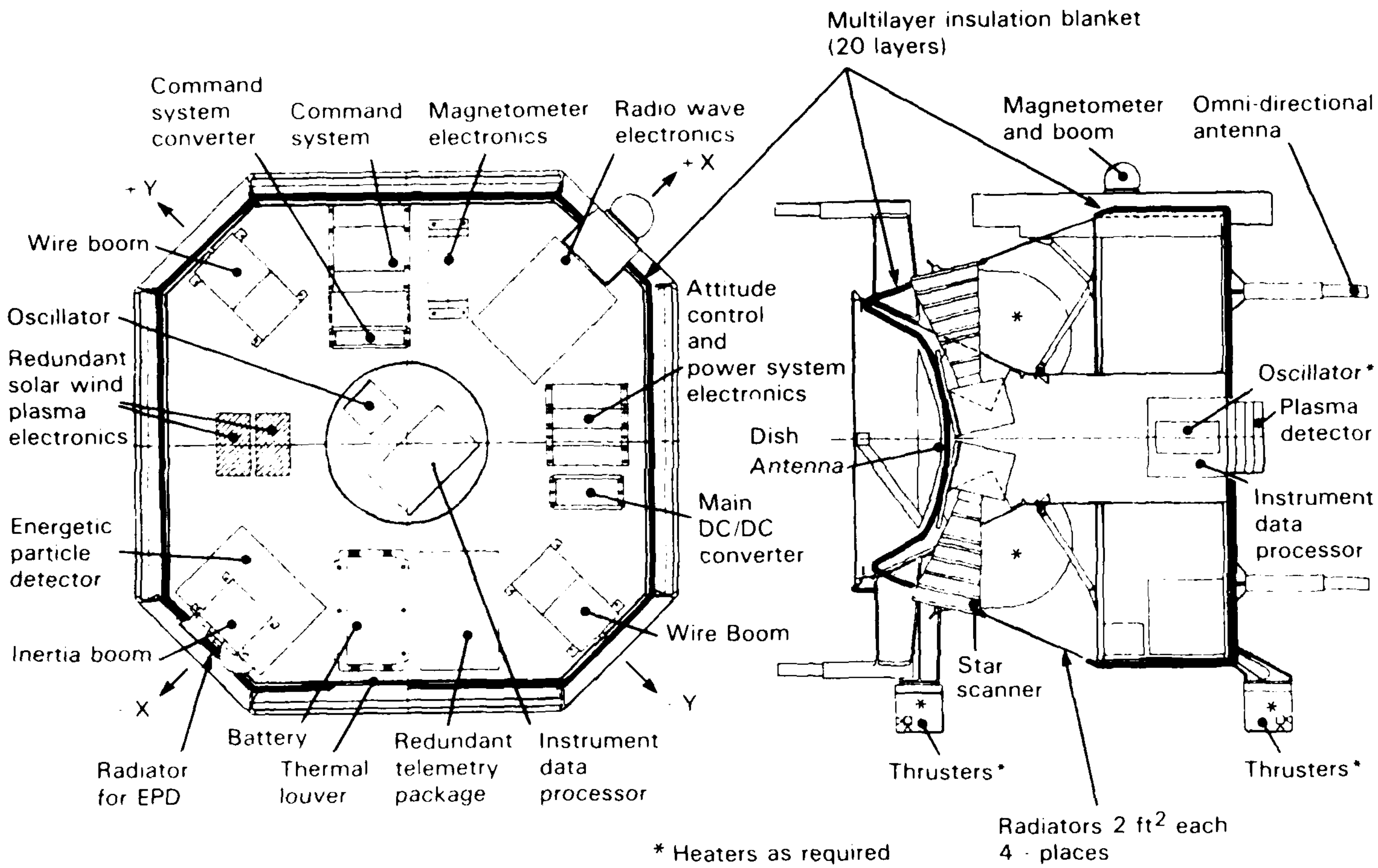


Fig. 6-16. **SIMPL thermal design schematic.**

lack of tape recorders, this is generally true for SIMPL. However, replacement heaters will be available in the event a system must be turned off.

Solar array temperatures will be nearly constant at about 50°C since the Sun angle variation is small and the panels can radiate from the back side. Long life should result from the lack of thermal working.

The propulsion system requires special consideration because the fuel freezes at 0°C. Because the tanks and fuel lines are largely contained in the temperature-controlled interior, these items will not require additional protection. High-powered RF components in the conical adapter section will provide heat for the fuel tanks. Heaters as required will be applied to the external fuel lines and thruster assemblies.

Some instruments have unique requirements and require special consideration. The particle instrument requires temperatures of from -20 to -10°C. This requirement is achieved by a dedicated radiator and thermal isolation for the head, which is the critical part. The boom-mounted magnetometer sensor heads have a wide temperature tolerance and can easily be handled with passive techniques. The other instruments have requirements similar to the other electronic components.

There is some concern about the effects of the hydrazine thruster plumes on the external insulation surfaces. At this point little is known about the flux distribution or the intensity. However, it is certain to be much less severe than AMPTE's solid rocket firing environment and can probably be handled with local shielding of heavier Kapton.

During the preliminary design phase the basically passive design of SIMPL will be analyzed in greater detail. The need for a battery louver and the size and location of the thruster and other heaters will be determined, and the overall feasibility of the design concept will be established.

6.8 RF COMMUNICATIONS AND RANGING

S-band transponders compatible with the Air Force Space-Ground Link Subsystem (SGLS) will be used to telemeter the instrument data and spacecraft housekeeping, command the spacecraft, and determine the range and range rate. During normal operation the telemetry will be collected continuously and in real time by four receiving stations to be colocated with SOON/RSTN stations. Operation of the spacecraft (command and ranging) will be by the Air Force Satellite Control Facility (AFSCF).

The uplink signal is in the 1.76 to 1.84 GHz frequency band. Command signals, using the standard SGLS ternary FSK format, are phase modulated onto the carrier at 1000 bps. In addition, a 1 Mcps PRN ranging code can be uplinked. This code is retransmitted by the transponder and received by the remote SCF ground station. The received code is correlated with a delayed replica of the transmitted code to determine range. In addition, the spacecraft transmitter frequency is coherently locked to the received uplink frequency. The frequency difference as received at the ground allows range rate to be determined. A typical uplink spectrum is shown in Fig. 6-17.

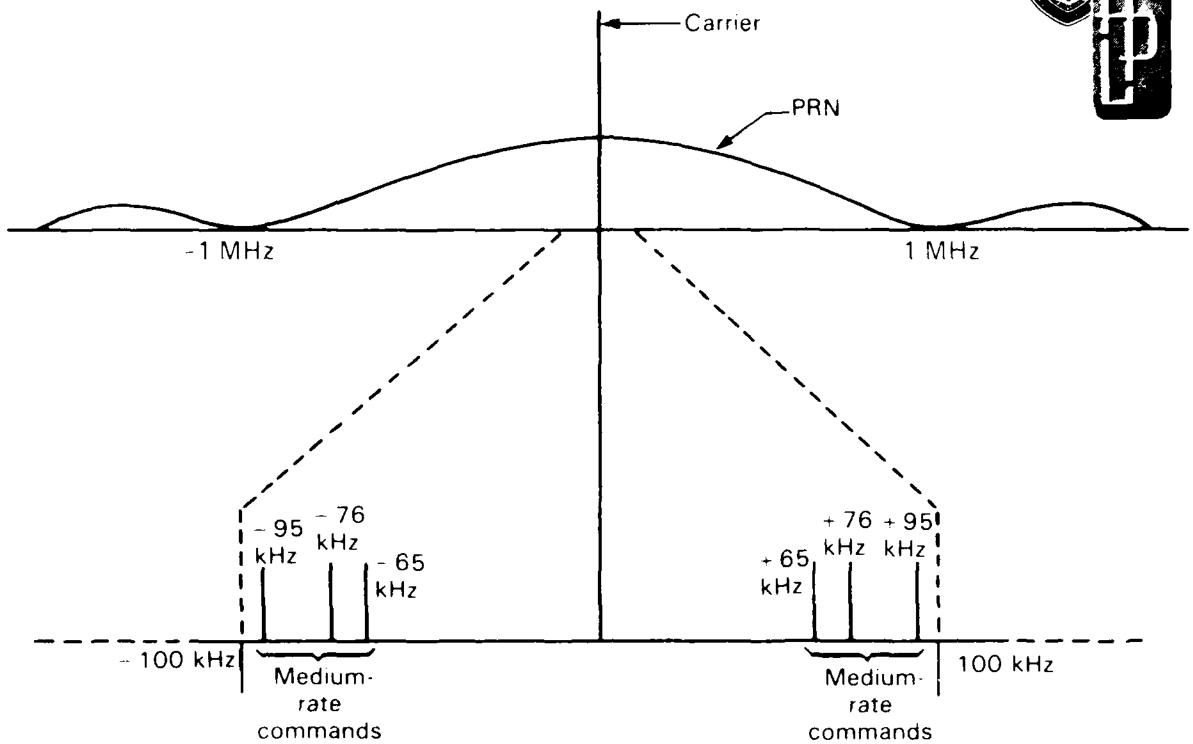


Fig. 6-17. SIMPL uplink spectrum.

SIMPL will utilize only SGLS downlink Carrier 1; Carrier 2 and the 1.7 MHz subcarrier will not be needed. Carrier 1 is at a frequency of 256/205 times the uplink frequency, either coherently locked or crystal controlled. The transponded PRN ranging code is directly modulated onto this carrier. SIMPL's telemetry has a single rate of 100 bps. This will be convolutionally encoded (rate 1/2, constraint length 7) and the resulting 200 symbols/sec will modulate the 1.024 MHz subcarrier. The downlink spectrum is shown in Fig. 6-18.

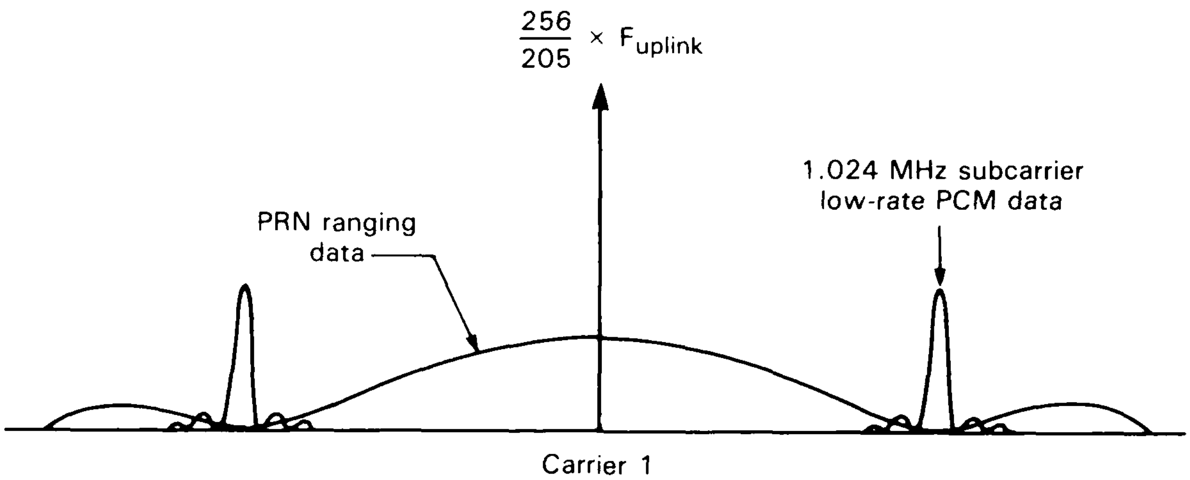


Fig. 6-18. SIMPL downlink spectrum.

As explained in Section 4.3, when the spacecraft is on station and stabilized, the line of sight between the spacecraft and Earth is maintained within 10° of the spin axis. If another 1° is added to account for attitude control error, then antenna coverage is required only to ±11° from the -Z axis. In this situation a "high gain" antenna will be used for both uplink and downlink.

A simple parabolic reflector antenna can perform this function reliably and at low cost. As the diameter of such an antenna is increased, the boresight gain increases and the beamwidth decreases. The gain at 11° is maximized at a diameter of about 3.8 wavelengths, as shown in Figure 6-19. Below this value the boresight gain is too low, while above that value the beam is too narrow. It is estimated that hardware losses and the

requirement for dual frequency operation will limit the gain of this antenna to 14.0 dBic at 11°; that figure has been used in the link analyses. Section 7 summarizes the link margins; the detailed link calculations are shown in Appendix D.

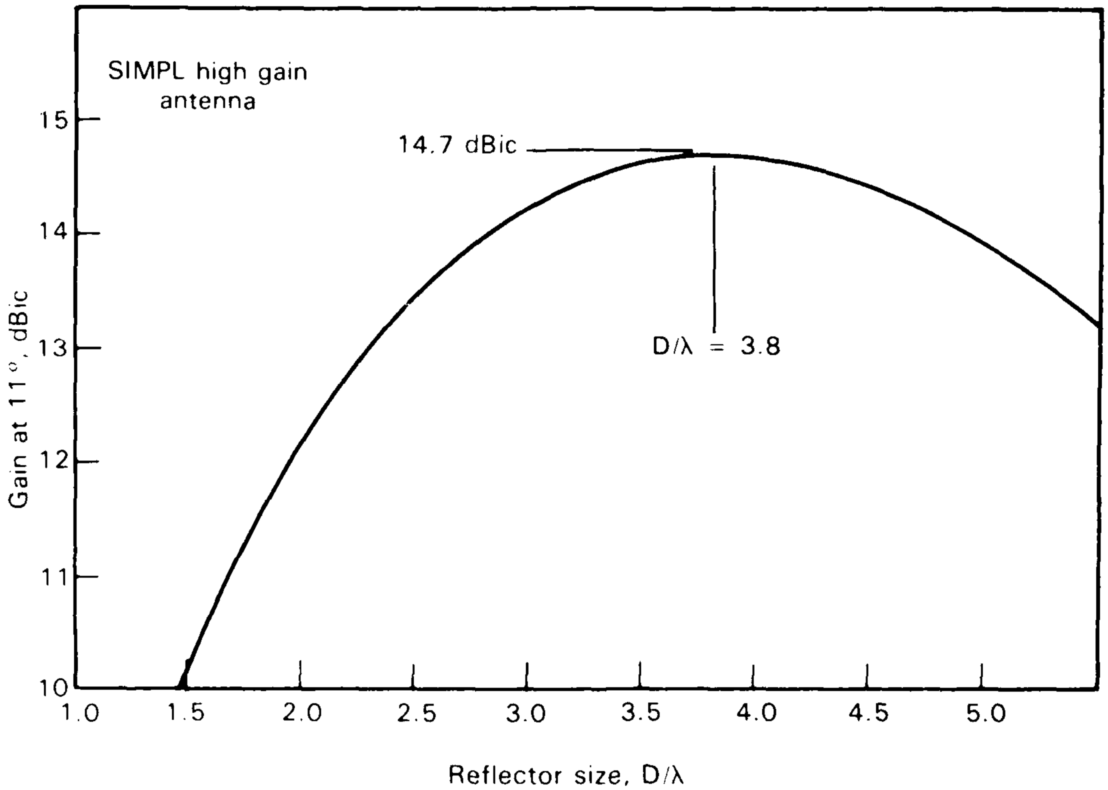


Fig. 6-19. SIMPL high gain antenna.

During the transfer orbit the line of sight may not be confined within the 11° cone. Also, provisions for communications must be made if excessive attitude errors are encountered while on station. A switched hemispherical antenna system similar to that flown on AMPTE has therefore been included. Resonant quadrifilar helix antennas provide RHCP and a nominal worst-case gain of -2 dBic over a hemisphere. Since these antennas are narrow band, and the SGLS coherence ratio is much wider than NASA's, separate transmit and receive helices will be used.

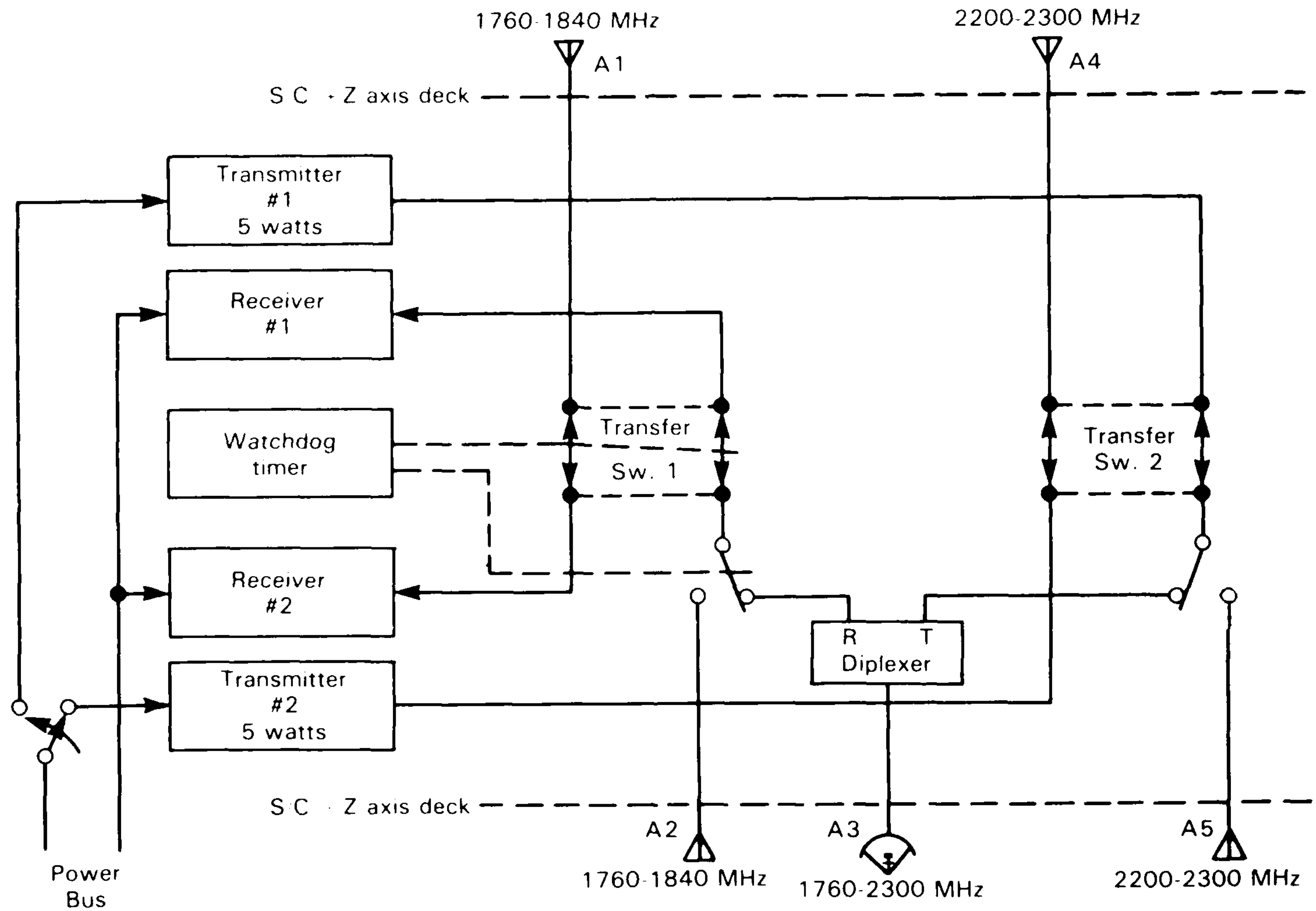
A block diagram of the RF communications system is shown in Fig. 6-20. Extensive redundancy is included. Two sets of helices cover the +Z and -Z hemispheres. A single reflector antenna is used with a diplexer for both transmit and receive within the 11° cone. Two transponders with 5 W transmitters are used. This output power can be obtained by substituting a TDRSS output stage for the standard SGLS stage. The 5 W transmitter will draw 28 W from the bus; that is about the highest practical transmitter power for a power system of this size. Transfer switches allow either transponder to connect with either set of antennas. To cover attitude emergencies, one transponder is always switched to the +Z antennas. Two SPDT switches are used to choose between the -Z helices and the high-gain antenna.

A watchdog timer has been included. If a command has not been received within a set length of time, the transponders are switched to the hemispherical antennas under the assumption that an attitude error has misdirected the high gain antenna. The watchdog can also switch the transponders to recover from a receiver failure.

6.9 COMMAND AND DATA HANDLING (C&DH) SYSTEM

The C&DH system must control and collect data from the various on-board instruments, systems, and data processors. Real-time encrypted downlinks must be compatible with both the SOON/RSTN and SCF stations. SGLS compatible authenticated commanding, both real-time and delayed, is required.

The C&DH design for SIMPL borrows heavily from our AMPTE and GEOSAT designs. Because C&DH is so critical to the mission, full redundancy is employed. Communication security concepts from



Antenna polarizations are RHC
A1, A2, A4, A5 are quadrifiliar helicies; BW = 180°, G ≥ − 2dBic
A3 is parabolic; BW = 18.3°, G ≥ 14dBic at 11°

Fig. 6-20. SIMPL RF communications system.

our GEOSAT spacecraft have been applied, but taking advantage of the latest spaceborne cryptographic equipment that will be available in time for SIMPL.

6.9.1 On-Board Processors

To provide the required command and data handling for the instruments on SIMPL, several on-board processors will be required. One will perform command execution, one will handle the collection and formatting of downlink data, one will process attitude sensor data to generate attitude vectors, and one will process instrument data to generate the required spacecraft data product.

Some general comments which apply to all of these processors is in order. First, even though they are shown as isolated processors in the block diagrams shown in Figs. 6-21 through 6-23, when the actual implementation of the hardware is done, some of these functions may be combined. At the present time the most likely possibility is the combining of the attitude and the Instrument Data Processor functions into a single Instrument Processor.

The Command and the Telemetry Processors will most likely be small special purpose processors with PROM based software which will make them into hardwired designs. For SIMPL, there appears to be little benefit in trying to combine these functions into a larger more powerful processor even if that processor could handle both functions. Keeping the Command and Telemetry Processors separate improves the fault tolerance of the spacecraft as a whole

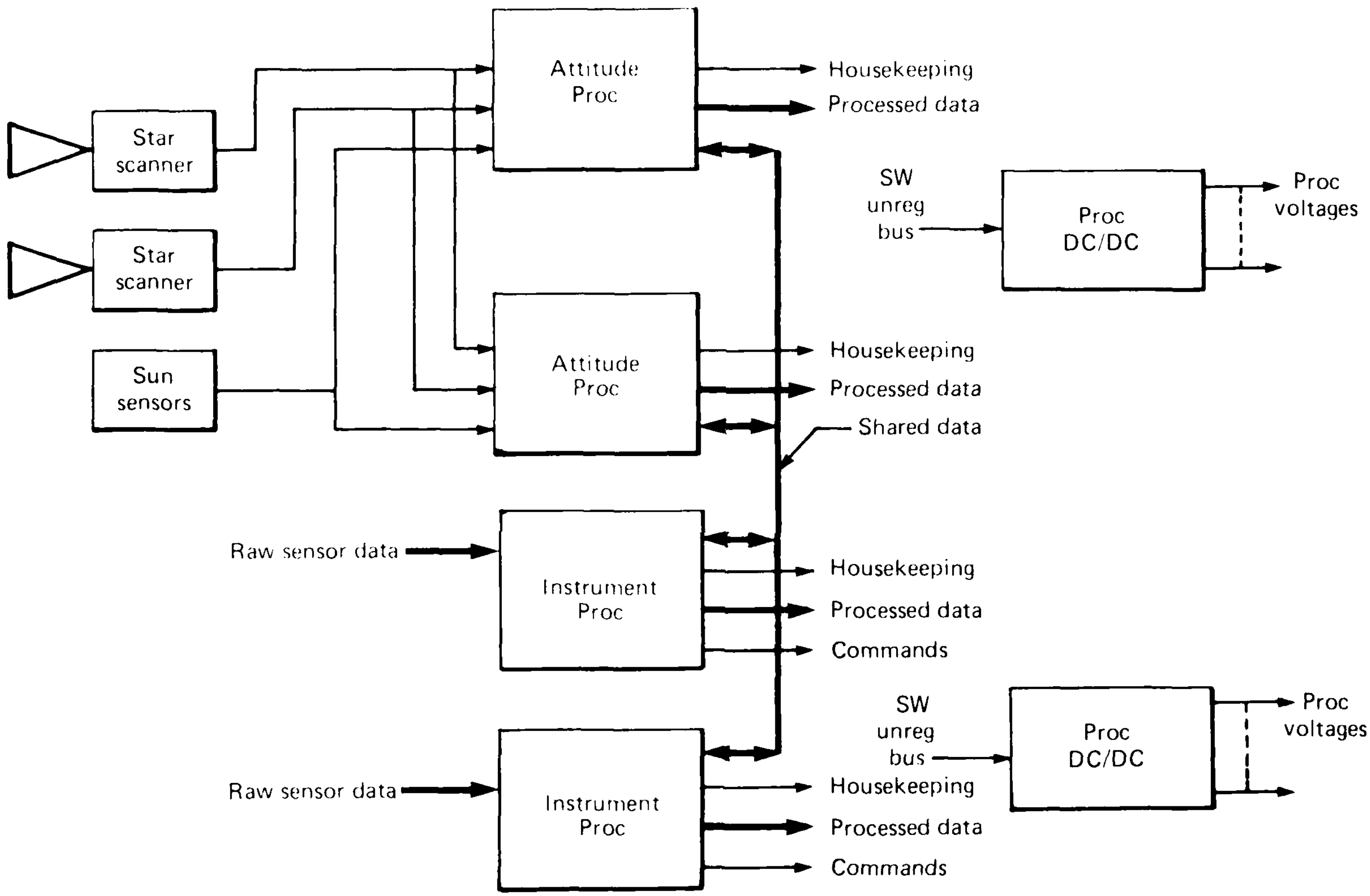


Fig. 6-21. **SIMPL instrument processor system.**

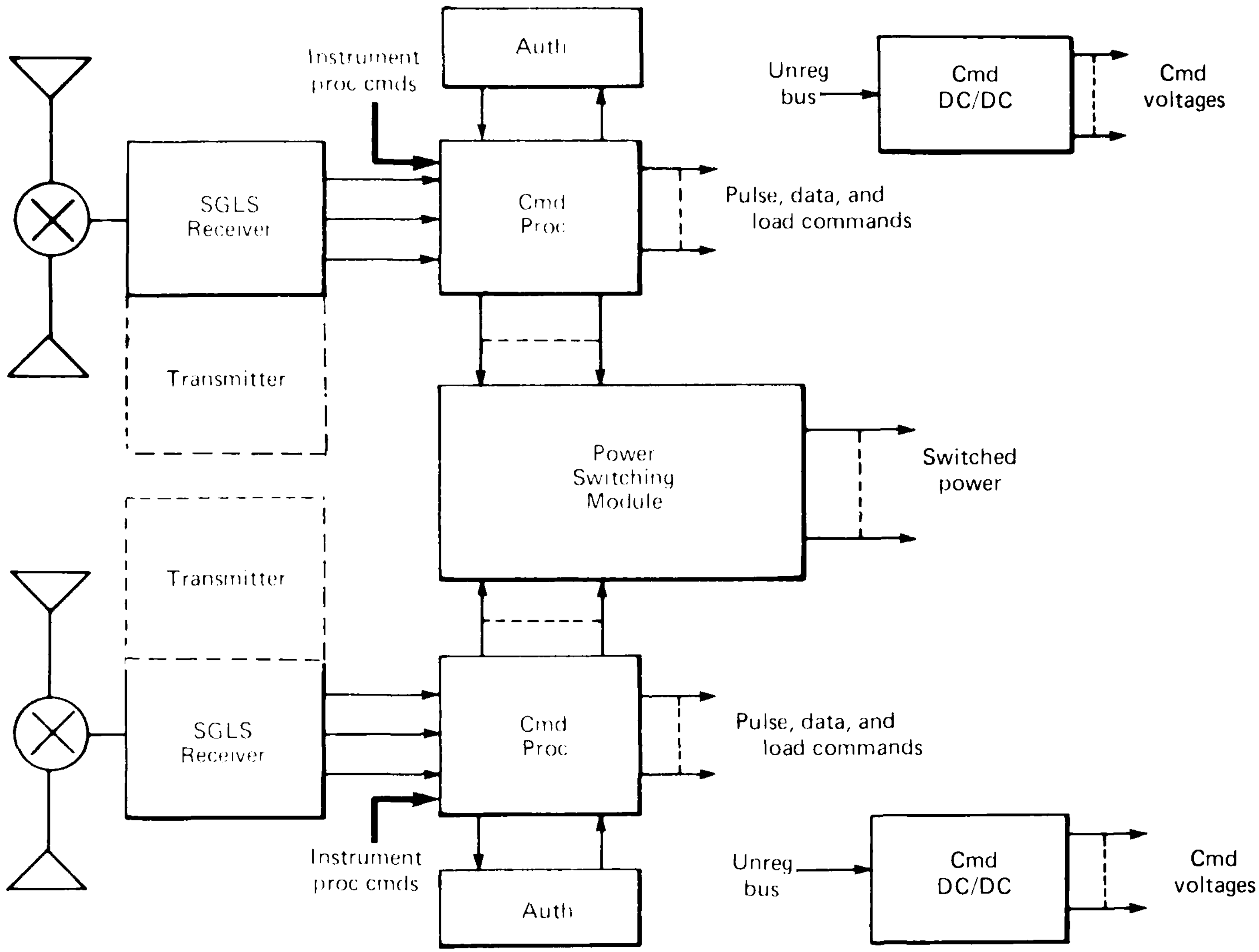


Fig. 6-22. **SIMPL command system.**

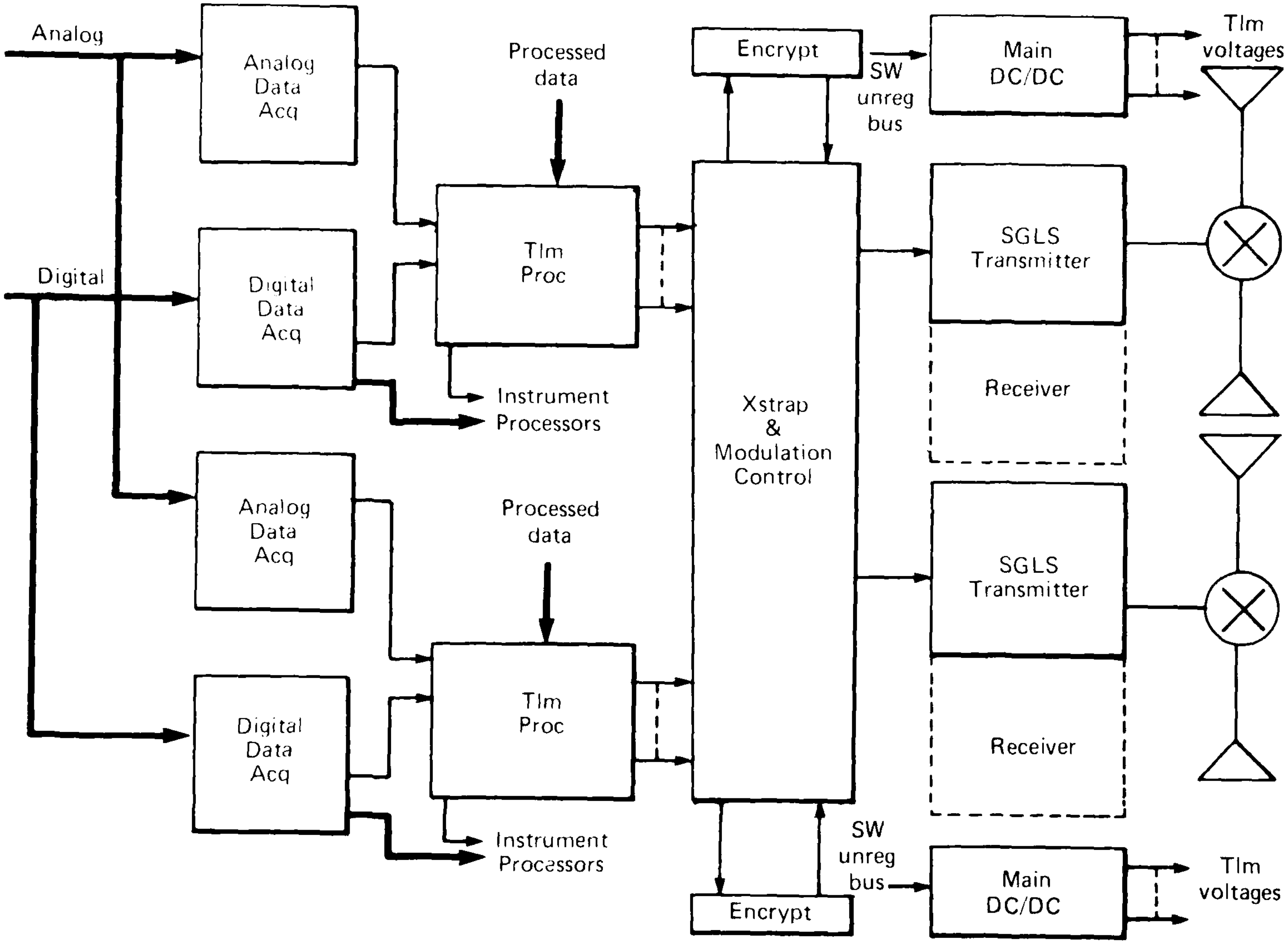


Fig. 6-23. SIMPL telemetry system.

by keeping faults from propagating into multiple functions. It also simplifies the coding and checkout of the separate processors. All of the processors on SIMPL will be implemented in a redundant fashion to support the six-year lifetime goal.

The hardware implementation of these processors will require some further study. The RCA 1802 8-bit CMOS microprocessor, which has been used by JHU/APL in almost all its spaceborne processor systems, is no longer available in a Hi-Rel version. The AMPTE and GEOSAT spacecraft designs on which SIMPL is based used this processor in several subsystems. The processor board in these systems will have to be redesigned to accommodate a new microprocessor chip set. At the present time, the most likely replacement is the Harris 80C85RH radiation hardened CMOS processor chip set. This 8-bit chip should more than meet the requirements for the Command and Telemetry Processors. While SIMPL's radiation environment is not nearly as severe as AMPTE's, moderate radiation hardness is required (see Appendix B). In addition, radiation hardened chips usually display superior resistance to single event upset (SEU). When SIMPL enters the preliminary design phase, this and any other available processors will be evaluated, and a hardware decision made.

For the spaceborne Instrument Processor, the hardware implementation is less clear. The exact requirements for the processing have not been defined, but the present estimate of the processing power required is felt to be no more than an IBM PC with an 8087 floating point co-processor. Currently, processing for similar instruments is done on large mainframes (up to a Cray 1); complete algorithms for a smaller machine do not yet exist. This area definitely needs additional study in the next phase of the program.

The IBM PC is a 16-bit machine based on the Intel 8088 microprocessor chip set. It has floating point capability (this appears to be an essential requirement) when used with its 8087 math coprocessor. This NMOS chip set is not available in a radiation hardened version at this time, but might still be usable if sufficient shielding were provided. Therefore, to maximize our options, we have placed the Instrument Processor in the spacecraft center column, potentially the most benign environment. A related unhardened NMOS processor, the 8085, was flown on GEOSAT as part of the primary instrument.

Other processors are available, and can be evaluated more thoroughly as SIMPL's processing requirements become better defined in the next phase. The recent introduction by Motorola and Hitachi of a CMOS version of the 68000 is the most recent [illegible] for SIMPL. Since new processors are being introduced [illegible], and Harris and Sandia are actively working on new radiation hardened designs, there may be several flightworthy 16-bit processors to choose from when SIMPL reaches its hardware phase. The APL routinely evaluates parts for total dose hardness in its own Co-60 chamber, and will soon be equipped for SEU testing. The table below lists some currently available 16-bit processors that are candidates for the Instrument Processor:

VENDOR	PRODUCT	HARDNESS (Rads)	COMMENTS
Fairchild	F9450	?	1750A
Intel	8086	1,000?	Has FP, may shield, not CMOS
RCA	EPIC	100,000	Bit slice, CMOS/SOS, no FP
TI	9989	1,000,000	IIL, no FP?

EMM	SECS-80	10,000	Boards, Intel chips
Mikros	MKS1750	200,000	Boards, CMOS/SOS, 1750A
Motorola	HD68HC000	?	Introduced 9/85

6.9.2 Command System

A block diagram of the command system is shown in Fig. 6-22. It is completely redundant at the package level up to the Power Switching Module (PSM), which contains the individual relays that switch power to all of the other spacecraft systems. Special techniques within the PSM (explained below) achieve nearly full redundancy within this unit as well.

The command system receives the command message from the receiver section of the SGLS compatible transponders, works with the NSA-supplied authenticator to provide command authentication, and processes the command data in the Command Processor. It provides outputs to all of the other spacecraft subsystems as digital signals or as switched power from the PSM. Both redundant halves are powered all the time from their own redundant DC/DC converters.

The command processor must implement a complement of commands similar to that of the AMPTE or the GEOSAT command processors that we recently designed, built, and launched. It will decode and process relay commands, logic level pulse and data commands, and "long load" commands to load data and programs into the RAM memory of other processors. It will also use a long load command itself to store uploaded command sequences for delayed execution. This processor will provide redundant outputs for all

decoded commands, and will switch relays in the partially redundant PSM using our flight-proven method of carrying the redundancy up through the coils of the relays.

The command rate and data format will conform to SGLS standards. It currently appears that the RF links can support a command rate of 1000 bps with no need for a second, lower rate. This rate, eight times higher than used for AMPTE, will facilitate processor reloading.

In addition to providing real-time execution of commands issued from the ground and a delayed command capability for stored sequences of commands, the command processor will have an input from the Instrument Processor to allow it to "close the loop" onboard the spacecraft. This implementation will allow maximum flexibility for using the processing power on the spacecraft, while maintaining a ground control override in the event of problems or failures. This capability was employed on both AMPTE and GEOSAT, and a similar implementation will be use on SIMPL.

One function which does not appear as a separate block in Fig. 6-22, but which may be implemented as part of the Command Processor, is the Time Management Unit (TMU). To provide time annotation of all of the telemetry data, a spacecraft clock (with time traceable to UT) will have to be maintained onboard. This time information is required in the downlink data product, in the attitude processing, and for delayed command execution timing. The TMU will be driven by the precision Dual Oscillator and will have a command interface for receiving drift corrections sent from the ground.

The Command Processor will also implement the Watchdog Timer functions required by the authentication and RF switching systems. Should no command be received within a preset period of time, the Timer will begin authenticator recovery procedures and also switch the command receivers to the omni-antennas.

6.9.3 Telemetry System

A block diagram of the SIMPL Telemetry System is shown in Fig. 6-23. It is a redundant system with each side consisting of the Analog and Digital Data Acquisition hardware, a Telemetry Processor, the Cross Strap and Modulation Control hardware, and an NSA-supplied encryption device. Power is provided by the redundant main DC/DC converter. All of this hardware will be connected to the transmitter portion of the SGLS compatible transponders. Unlike the command system, only one of the redundant telemetry systems will be powered at any time. However, either side can collect all of the telemetry data and send it to the ground through either transmitter.

The Telemetry Processor will control the acquisition of all instrument and housekeeping data from the spacecraft subsystems. It will format these data into fixed frames, attach synchronization codes, provide frame counts and timing information, and send the data to the Cross Strap hardware. Some of the raw instrument data collected by the Telemetry Processor are required by the Instrument Processor. The data are collected in the Telemetry System, scaled and A/D-converted if necessary, and routed to the Instrument Processors as a digital stream. Processed data (forecast parameters) are returned from the Instrument Processor, merged with the rest of the raw instrument data and the housekeeping data and formatted into a fixed frame.

A "memory trickle" function will be provided to return the contents of on-board processor memories, including the delayed command load. Because complete memory dumps will take many minutes, on-board memory error trapping will be used in addition.

Because of the distances involved, and the small receiving antennas at the SOON/RSTN stations, the downlink data rate is limited. The current baseline is 100 bps. The approximate breakdown of the data within this 100 bps allocation is shown in the table below. Note that only two modes are required for the telemetry data. These are fixed format modes; at this point we see no need for in-flight reprogrammability since the Instrument Processor will have this capability for the instrument data. Each of the two modes will have its own data format, and the header information in each frame will identify which type of data is in the frame. This will simplify data reduction at GWC/Omaha. In general, these mode changes will only be made by ground command, and the maintenance mode will only be used for transfer orbit operations and if needed for trouble-shooting once the spacecraft becomes operational.

	MODE	
DATA TYPE	SCIENCE	MAINTENANCE
Processed data	15 bps	---
Raw data	45 bps	---
Attitude data	10 bps	50 bps
Housekeeping	10 bps	20 bps
Memory trickle	10 bps	20 bps
Header, overhead	10 bps	10 bps
TOTAL	100 bps	100 bps

The entire data stream will be encrypted by the NSA encryption device, and there will be no plain text data link from the spacecraft. Note that the figures shown above are data rates, not the actual transmitted symbol rates. To achieve the required bit error rates at these distances to small receiving stations will require the use of error correcting coding. We have baselined the NASA standard rate-1/2, constraint length 7 convolutional code, which offers significant performance with minimal encoding hardware. In normal operation one transmitter will be on continuously, and the formatted telemetry data stream will be sent 24 hours/day.

The design of the data acquisition and the processor portions of the Telemetry System will be based on the AMPTE and GEOSAT designs recently flown. With the exception of the processor boards used, much of the hardware can use existing designs. The Cross Strap and Modulation Control portion will be based on the AMPTE design which provided this function for a NASA Standard Transponder, and which included a convolutional encoder. This design will have to be modified to provide the unique interface circuitry required by the SIMPL hardware systems, and will have to include interfaces for the encryption hardware.

6.9.4 C&DH Make or Buy

A number of commercial vendors have command, telemetry, and processor systems available "off the shelf." There is a potential for some cost savings to the program if some of these components can be used for SIMPL. JHU/APL briefly surveyed SGLS compatible units available from Gulton, Cincinnati Electronics, and several other suppliers. Most of these units had some deficiency of importance to SIMPL, such as inability to operate in authenticate mode, lack of delayed commands, lack of long load

capability, lack of a Watchdog Timer, or excessive power. When the missing features and customized interfaces are added, apparent savings sometimes vanish. For example, the PSM would almost certainly have to be customized for SIMPL.

However, when the requirements for this hardware become better defined in the next phase, the available commercial equipment will again be surveyed and evaluated for possible use. Meanwhile, to provide the most conservative cost estimate, the designs shown in this report assume hardware designed and built at JHU/APL using as much of the existing AMPTE and GEOSAT hardware design as is practicable.

6.9.5 Communications Security (COMSEC)

Authenticated commanding and downlink encryption are program requirements. JHU/APL has experience with both these COMSEC techniques, most recently for our GEOSAT-A satellite for the Navy. The GEOSAT-A requirements were more difficult than SIMPL's since we had to synthesize our own authenticator function using NSA-supplied decryptors, and the downlink involved tape recorded data as well as some plaintext. GEOSAT-A carried four KG-46 encryptor-decryptors, each weighing 3 lb and using 4 W when powered.

In discussing SIMPL's COMSEC requirements with NSA, two new encryption and authentication developments were identified. Smaller and lighter than the devices used on GEOSAT, these NSA-developed chip sets will perform the encrypt and the decrypt/authenticate functions, respectively. Supplied on single cards, they will integrate directly into our C&DH hardware. The weight,

The radio experiment, which covers the frequency range from 10 kHz to 1 MHz, will be sensitive to all of the DC/DC converters in the spacecraft. To minimize this interference, all the converters will operate at high frequency (~50-80 kHz) and will be either synchronized or crystal controlled. They will thus represent a single frequency source with widely spaced harmonics.

The magnetometer will encounter an average interplanetary magnetic field of ~5 nanoteslas. Since this is a vector measurement, however, field orientations will often reduce the field strength along one of the axes to less than 0.1 nanotesla. A total stray field design limit of 0.1 nanotesla will prevent serious errors in the measured vector field direction. SIMPL will have an overall magnetics control plan with controlled stray fields from each box or subsystem. These limits will define both the depermed residual fields and the residuals following exposure to a three gauss perming field.

6.11 GROUND SUPPORT AND LAUNCH SUPPORT EQUIPMENT

Ground Support Equipment (GSE) and Launch Support Equipment are needed to control and monitor the SIMPL spacecraft during all of its test phases from initial integration through launch. Because a Shuttle launch is baselined, this must also include equipment designed to fly onboard the Shuttle to support the spacecraft before it is deployed from the cargo area. The requirements, a development approach, interfaces, and special considerations for a Shuttle launch are discussed below.

6.11.1 Support Equipment Requirements

Figure 6-24 shows the GSE block diagram. Specific features are listed below.

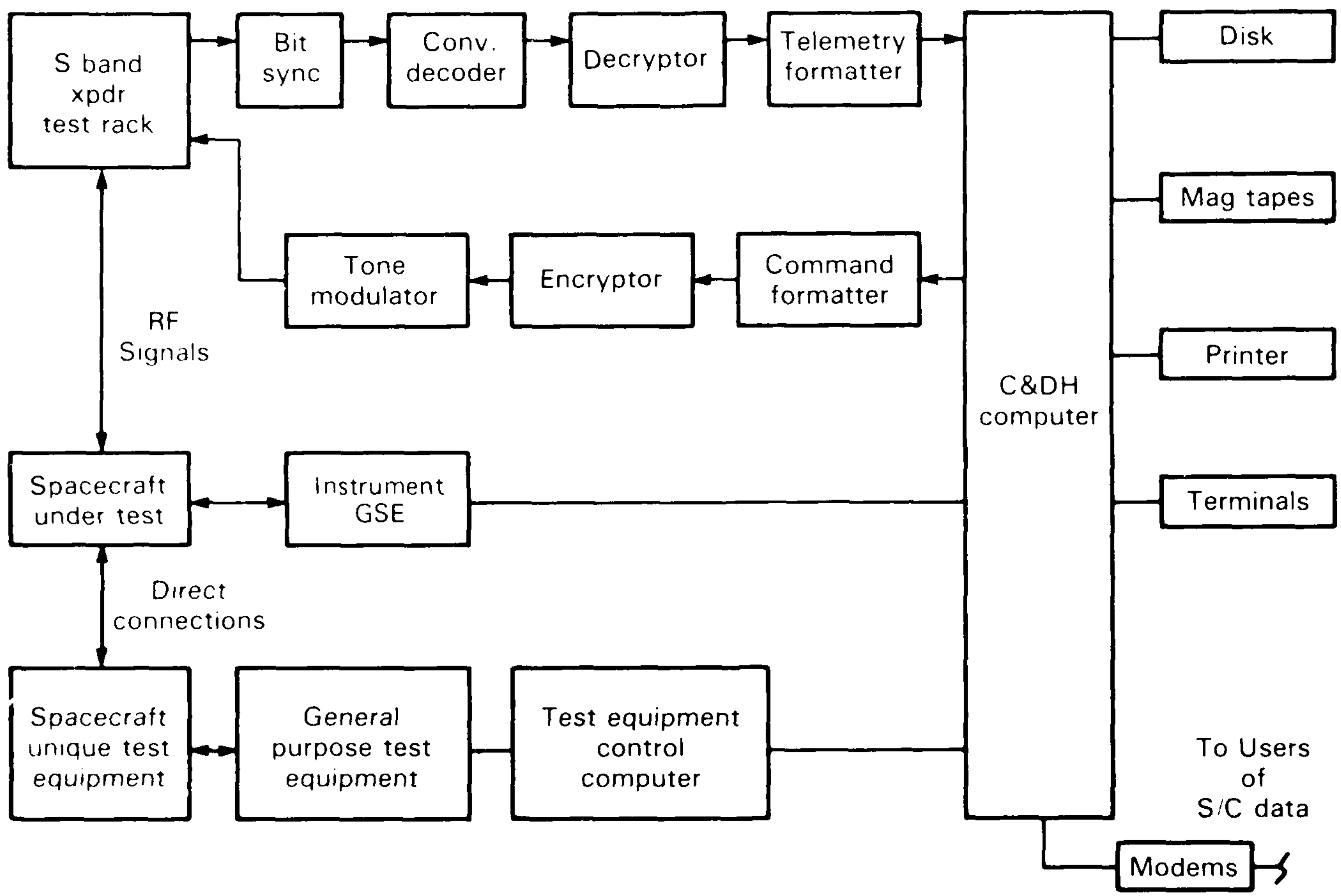


Fig. 6-24. **SIMPL ground support equipment for the spacecraft test configuration.**

1. Authenticated command messages are transmitted to the spacecraft and verified by echo checking through telemetry.

2. Telemetry data are demodulated, decoded, decrypted, decommutated, converted to engineering units, and displayed in real time.

3. Limit checks are performed on telemetry data.

4. All test data are logged on digital magnetic tape.

5. Printed copies of test data are available as needed.

6. A test procedure language is provided to facilitate sending commands, calling up displays, designating telemetry limit settings, etc.

7. Interfaces are provided for communication between the spacecraft GSE and instrument GSEs as required.

8. Post-test analysis of tape recorded data is provided.

9. Provisions are made for compatability tests with the SCF ground control network.

6.11.2 GSE Development Approach

The GSE computer interfaces and software must be developed early in the program for testing the command and telemetry systems. Following installation of the command and telemetry packages into the spacecraft, the GSE will be

reconfigured as shown in Fig. 6-24 and used for integrated spacecraft testing. Eventually the GSE will be shipped to the launch site with the spacecraft where it will continue to support spacecraft tests before and after mating with the Shuttle. After launch, control of the spacecraft will be transferred to the Mission Control Center (MCC). The GSE is then no longer needed in the field and can be returned to JHU/APL to support integration of the backup spacecraft.

6.11.3 GSE Functional Description

The spacecraft uses a coherent S-Band transponder for command, telemetry and ranging. A rack of RF test equipment, shown in Fig. 6-24, interfaces the S-Band signals to telemetry processing and command generation equipment.

Telemetry is bit synchronized, convolutionally decoded, decrypted, and frame formatted before being passed to the Command and Data Handling (C&DH) computer. The computer further processes the telemetry in real time for display on CRT terminals, printed output, logging on magnetic tape, and limit checking.

Commands are initiated by the C&DH computer, converted to serial bit strings by the Command Formatter, encrypted, and modulated into SGLS-compatible ternary FSK format before being sent to the S-Band Transponder Test Rack.

Direct connections to the spacecraft are required for many purposes during testing. These include reading nontelemetered data, simulating solar array currents, charging the spacecraft battery, and injection of sensor test stimuli. Spacecraft Unique Test Equipment, shown in Fig. 6-24, implements harness interconnections, signal conditioning, and other

functions. General Purpose Test Equipment, such as voltmeters, spectrum analyzers, and counters, are also shown as an element of the block diagram (Fig. 6-24). These are interfaced to a Test Equipment Control Computer. The C&DH computer has a link with it so that the control of the overall test sequence and access to all test data are maintained in the C&DH computer.

Instrument GSEs are included in Fig. 6-24 to show that they can be given access to both the spacecraft and the data within the C&DH computer. Instrument data can be stripped out by the C&DH computer and fed to the instrument GSEs so that instrument testing can appear to be an extension of bench testing. In this way we can take advantage of specialized instrument GSE capabilities without having to duplicate them in the spacecraft GSE.

A modem interface is shown in the block diagram for communication with other spacecraft data users. SIMPL's low telemetry data rate will make it readily adaptable to telephone line interfacing techniques. This path will probably be the one used to perform compatability tests with the SCF ground control network. The same path can be used to communicate with remote instrument teams, so they need not be present at all times during integration.

6.11.4 GSE Questions Needing Further Study

A selection must be made of the computers and other major equipment for the GSE. The GSE system that was used for AMPTE has sufficient capability for the tasks outlined above for SIMPL. The availability and cost of maintenance support for the components of

this system in the SIMPL time frame will have to be determined. Depending on feasibility, portions of the AMPTE GSE components may be useful in the SIMPL GSE.

Network compatability tests need to be defined in such a way that they can be performed without interfering with the spacecraft test schedule. Sponsor requirements on the scheduling of the backup spacecraft will determine whether or not a complete second GSE will be needed.

6.11.5 Airborne Support Equipment (ASE) for Shuttle Launch

Because a Shuttle launch has been designated, Airborne Support Equipment (ASE) will be required. The spacecraft ASE is carried aloft on the PAM-D cradle and supports the spacecraft prior to deployment. It is responsible for payload safety and power conditioning, and can also provide any desired level of spacecraft pre-deployment health checking via command, telemetry, and the Orbiter avionics. Certain "baseline" services are provided to every PAM-D payload. These include a dual fused power bus, discrete commands, analog and discrete monitor points, access to the T-0 umbilical for pre-launch tests, and a path for transmission of payload telemetry to the ground via the Payload Data Interleaver and Orbiter RF links. Figure 6-25 shows the payload spacecraft, PAM-D, ASEs, and Orbiter avionics used to accomplish these functions. Slip rings are used to pass signals and power between the stationary ASEs and the spinning PAM-D and its payload.

Additional "mission specific" capabilities can be implemented in the Electronic Control Assembly shown in Fig. 6-25. These might typically include staging signals to indicate PAM-D launch status to electronics in the payload.

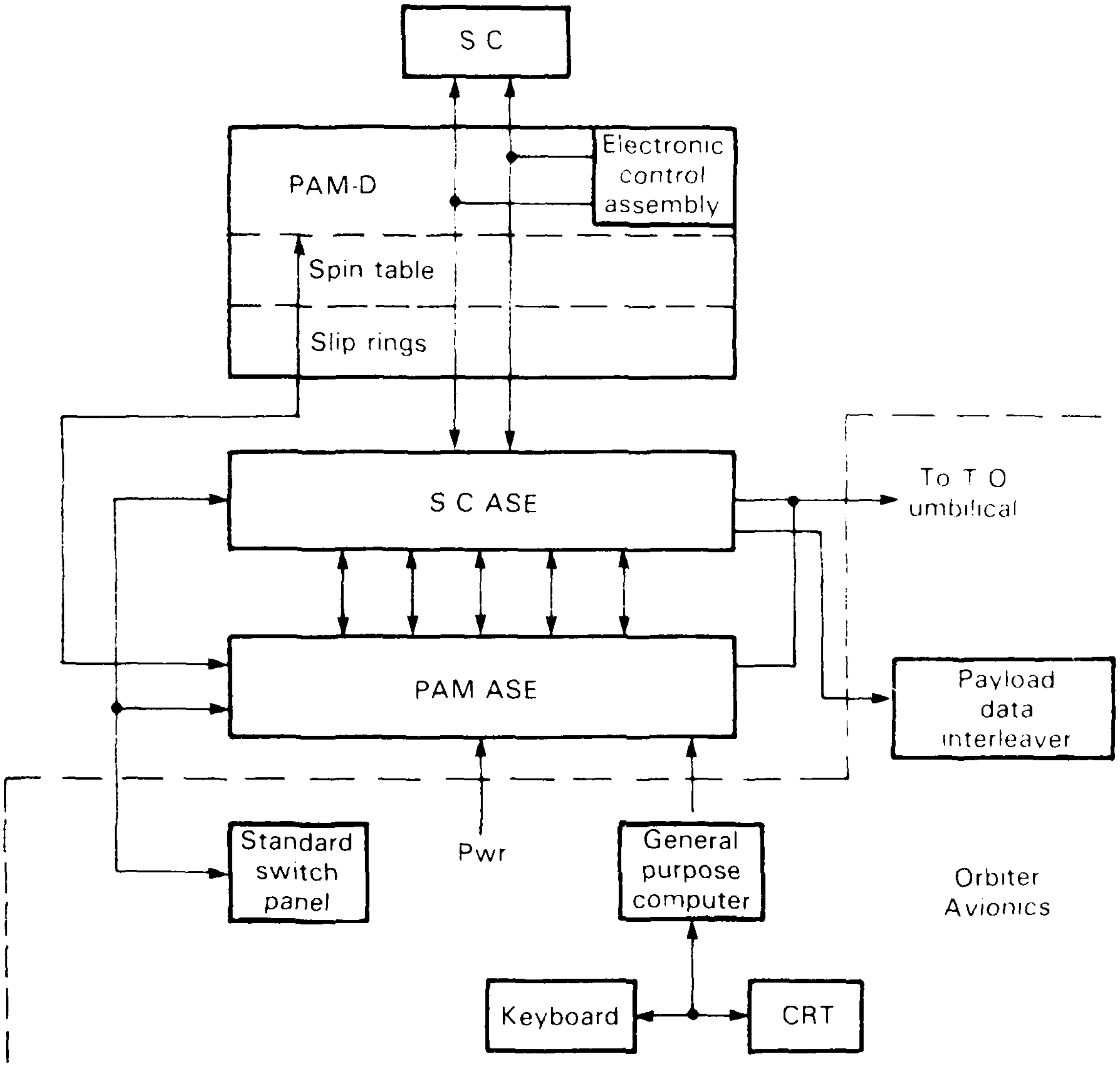


Fig. 6-25 SIMPL airborne support equipment (ASE) block diagram.

Further examination of the ASE requirements will be needed for the SIMPL program. However, JHU/APL studied the desirability of pre-deployment health testing during our WIND spacecraft design for NASA. Based on that experience, we expect to recommend a minimal amount of post-launch pre-deployment checkout for SIMPL.

Another function of the ASE is to provide contamination control, if needed, for protection of delicate instrument sensors while in the Shuttle cargo bay. JHU/APL studied dry gas purge requirements for the WIND sensors (channeltrons, microchannel plates, and solid state detectors). The experience gained from that investigation will help us design contamination control for the somewhat less sensitive Solar Wind and Particle instrument sensors on SIMPL.

7. GROUND FACILITIES AND OPERATIONS

Figure 7-1 is a block digram of the data flow for SIMPL. Collection of the spacecraft data product is performed 24 hours/day by dedicated equipment colocated at the four SOON/RSTN sites. These stations return the ciphertext (CT) data product in near real time via military packet network to Gl bal Weather Central (GWC) in Omaha. There, the data are decrypted and the forecast parameters extracted. GWC also serves as the central node for 24 hour/day collection of spacecraft health data, which is contained in the single downlink data stream.

Spacecraft command and control are entirely under the direction of the Air Force Satellite Control Facility (SCF), which also houses the SIMPL Mission Control Complex (MCC). The MCC receives ranging and telemetry data from, and issues command messages to, the seven remote SCF sites located worldwide. SCF contacts with SIMPL can be limited to a few brief periods each day since GWC is already receiving continuous spacecraft health data. A direct link from GWC to Sunnyvale provides health data to the MCC. The unusually low SIMPL data rate facilitates ground data transfer.

The SCF must also compute the orbit and definitive attitude so as to plan and execute spacecraft maneuvers. It also provides alerts and antenna pointing information to GWC, which then disseminates it to the SOON/RSTN sites.

Because of the requirement for authenticated commanding and encrypted downlink, many of the data paths will be carrying ciphertext. These links will require superior bit error rate performance to minimize effects of error extension due to encryption.

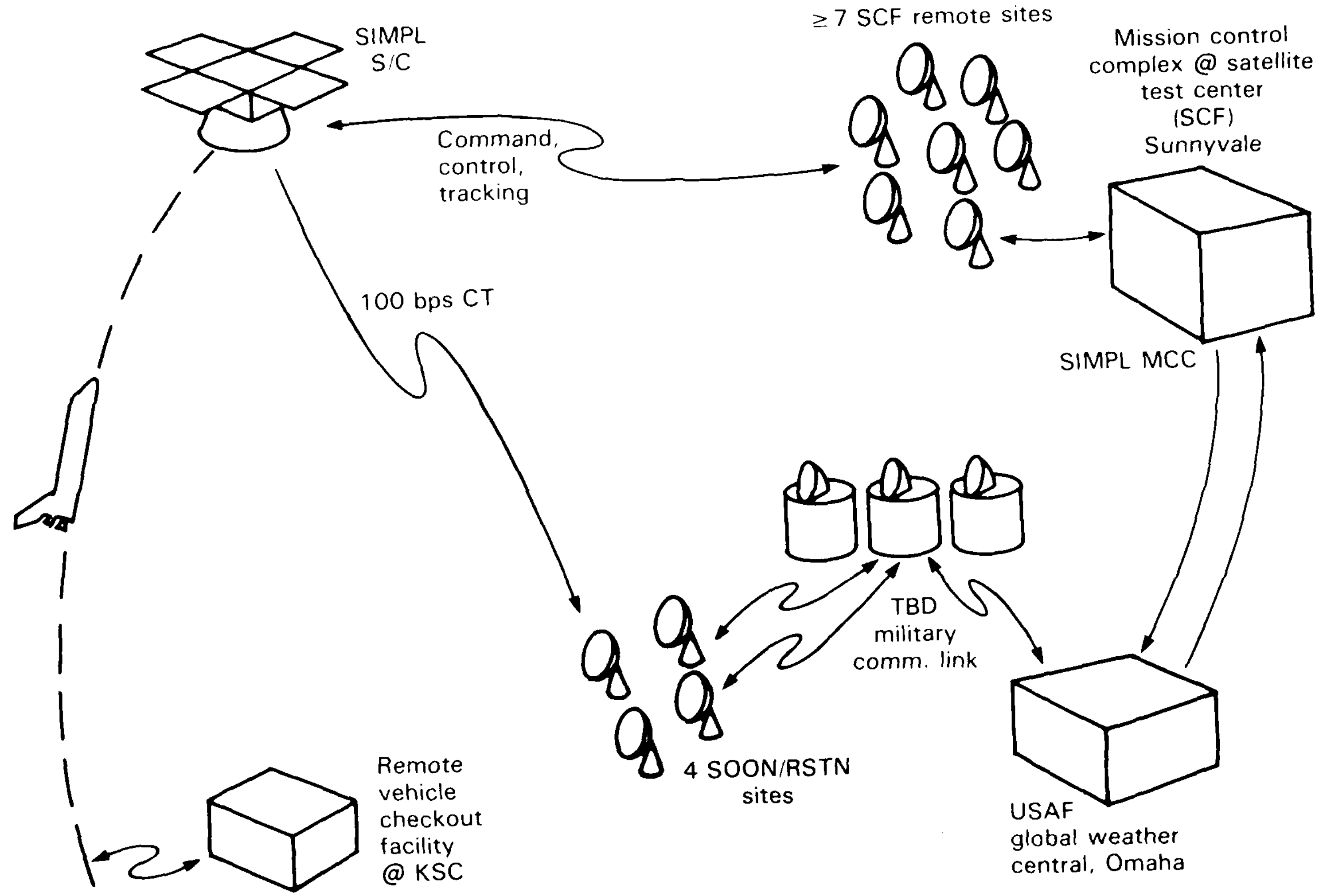


Fig. 7-1. **SIMPL data flow.**

SCF compatibility tests will be conducted prior to shipping SIMPL to the launch site (ESMC). There, SCF's Remote Vehicle Checkout Facility will support pre-launch checkout.

7.1 SOON/RSTN Station Design

This section describes the conceptual design of the SOON/RSTN stations. The four SOON/RSTN sites assumed for this study are:

	Learmonth, Australia	22°S,	114°E
	San Vito, Italy	40.7°N,	17.7°E
	Ramey AFB, Puerto Rico	18.5°N,	67.2°W
and either	Pelehua, Hawaii	21°N,	158°W
or	Castle AFB, California	37.3°N,	120.4°W

These stations will not be involved in the command and control of the satellite beyond their role of round-the-clock collection of both instrument and satellite housekeeping. No decryption or display of the data will take place at the local SOON/RSTN sites; they will act solely as a "bent pipe" for the encrypted data. This decision will limit SOON/RSTN software development to the bare minimum: antenna pointing, equipment control and status, and blocking and time-tagging data for recording and transmission. Figure 7-2 is a block diagram of the station.

To minimize costs we have designed the downlink to operate with modest receiving equipment at the SOON/RSTN sites. Each station will have a dedicated 5 m diameter antenna. A conventional 60°K noise temperature low noise amplifier (LNA) is assumed. Allowance has been made for the additional Sun noise

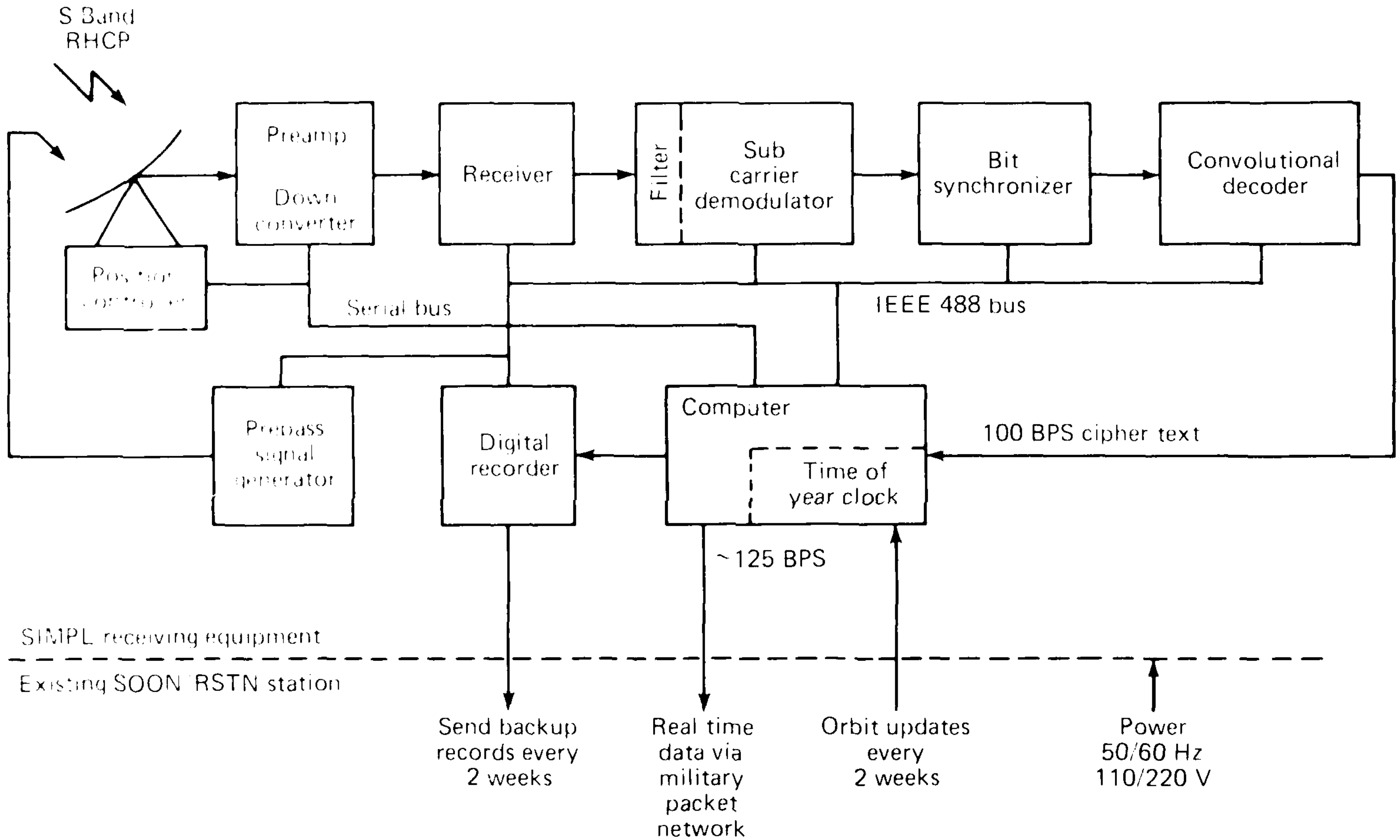


Fig. 7-2 Block diagram of SIMPL SOON/RSTN station.

caused by operating within 5° of the Sun (see Appendix E). The resulting system noise temperature (worst case) is 226°K. As shown in the RF link calculations (Appendix D), this data link from the spacecraft high gain antenna will have a margin of +8.4 dB.

An initial study by Scientific Atlanta indicates that our excess Sun noise allowance can be achieved without a special feed design. To control costs, no radome has been assumed. The antenna will have to be stowed during violent weather, resulting in a small, but acceptable loss of data.

Due to the slow movement of SIMPL relative to the Sun (period nearly six months), an open loop tracking system will suffice. No autotracking feed will be required, resulting in lower cost, improved gain, and better sidelobe control to reject Sun noise.

Pointing information will be generated by the station computer based on alerts received from GWC and will be sent to the position controller via serial bus. Use of a serial bus in this case will permit reasonable separations between antenna and computer. Synchro data will return by the same route. The LNA and downconverter will be integrated with the antenna. The downconverter translates the 2200-2300 MHz telemetry band to P-band (215-315 MHz). Any control signals to these units and any status information will also flow on the serial bus.

The next step in the reception process is the recovery of the 1.024 MHz SGLS subcarrier. The receiver to do this will probably be located in the SOON/RSTN building, although some designs might locate the receiver with the antenna.

A subcarrier demodulator is used to obtain the baseband signal. When the turnaround ranging channel is on in the spacecraft, uplink noise and ranging signals will also be present. Additional input filtering may be required to reject these signals, which are not processed at the SOON/RSTN stations.

A bit synchronizer receives the baseband signal and recovers the convolutionally encoded ciphertext, at a "symbol" rate of 200 sps. A convolutional decoder then outputs the original 100 bps ciphertext and routes it to the computer. Because the SIMPL bit rate is so low, it may be preferable to perform the convolutional decoding within the computer by software. That might be an advantage since multiple stations are involved.

The station computer controls the receiver, subcarrier demodulator, bit synchronizer, and convolutional decoder via an IEEE-488 bus. Status information from these units is also collected by the computer from this bus.

The 100 bps ciphertext is blocked and time tagged by the computer. Information concerning station status is also added, resulting in a loaded bit rate of about 125 bps. This data stream is formatted for the Military Packet Network and forwarded to GWC for decryption and analysis. As an example, five minutes worth of (loaded) SIMPL data can be sent in four seconds at 9600 bps. The blocking and packing for this example would, of course, introduce a five minute delay.

As a backup to the real-time data relay, the ciphertext and station status are also formatted for recording. These records will be sent to GWC every two weeks. A two week record of

data plus overhead would need less than 20 Mbytes, well within the range of today's low cost tape cassette technology. Control and monitoring of the digital recorder are also via the IEEE-488 bus.

The final package connected to that bus is a prepass signal generator. Prior to receiving SIMPL telemetry, a known S-band signal is injected into the antenna by a dipole at the vertex of the reflector. This signal is processed by the entire system, and an end-to-end readiness check is performed. Malfunctions can be detected and located in this way. Although "hot spare" redundancy is not called for, a full set of spare equipment will be stocked at each station.

Orbit updates will be supplied to the station at two week intervals using existing communications. This information can be entered manually, or on tape or floppy disk. No data line into the computer will be required. A relative-to-Sun-center coordinate system will probably be best for locating SIMPL, since the sites will already have excellent Sun location data.

The antenna assembly can be located on a concrete pad adjacent to the existing SOON/RSTN facilities. The rest of the equipment will be contained in the existing buildings, sharing their environmental control, AC power, and lightning protection. All the SIMPL equipment will be portable enough for economical dismantling in case of station relocation.

The station will be designed to run essentially unattended once a pass is started. Under normal conditions, an average of no more than 0.3 person will need to be budgeted per station for the SIMPL data reception.

7.2 AIR FORCE SATELLITE CONTROL FACILITY

The command and control of the SIMPL spacecraft will be the responsibility of the Air Force Satellite Control Facility (SCF). SIMPL must be guided through the transfer trajectory to be placed on station in orbit about L_1. Maneuvers must also be made periodically during the mission to maintain the orbit and to keep the high-gain antenna pointed toward Earth. The operation of this interplanetary spacecraft will place unusual burdens on the SCF, which normally deals with Earth-orbiting vehicles. Some facility improvements will be required to support the mission.

The orbit determination of SIMPL will be made from the range and range rate data collected at the remote tracking stations. As examined in Section 7.4, two or three 15 minute ranging passes per day will be sufficient when the spacecraft is on station. One of these passes should be from the northern hemisphere and one from the southern hemisphere. To improve the ranging link, it is suggested that the turnaround channel not be used when the spacecraft is being commanded, i.e., no simultaneous command and ranging.

Attitude determination is to be made from Sun sensor and star scanner data. As discussed in Section 6.4.1.2, these data may be processed onboard the spacecraft. The attitude estimate (or partially processed sensor data if full processing is not possible) will be downlinked with other housekeeping telemetry in the same stream as the instrument data. All telemetry can be received by the remote tracking stations. During normal operating conditions, the telemetry will also be continuously received by the SOON/RSTN stations and relayed to GWC. The housekeeping information should therefore be available from this source on a near real-time basis.

The orbit and attitude data will be used to plan the spacecraft maneuvers. Precessional torques will be applied at intervals ranging from 11 to 31 days to keep the high-gain antenna pointing toward Earth. Orbit correction maneuvers will be needed every one to three months. The orbit and attitude determination and the analysis leading to the commands to execute these actions will require a much more highly automated operation than was needed for ISEE-3, for example. This aspect of SCF support for SIMPL might be greater than that for most Earth-orbiting satellites.

It is estimated that the routine commanding of SIMPL can be accomplished by adding 10 minute time slots to the ranging passes of the remote tracking stations. Telemetry can be received during both uplinking operations. Once the operational orbit is achieved, support required from the SCF stations might therefore consist of as little as three 25 minute passes per day, plus pre-pass and post-pass setup times. The SCF may also occasionally have to cover data reception for an out-of-service SOON/RSTN station (during a hurricane, for example).

7.3 AFSCF LINKS

When in orbit about L_1, the SIMPL spacecraft will be at a range of approximately 267 Earth radii. The enormous path loss associated with this range makes the establishment of RF communication links more difficult than for closer Earth-orbiting satellites. The use of a fairly high-gain (+14 dBic) spacecraft antenna improves the situation. However, should an attitude error misdirect this antenna, the much lower gain (-2 dBic) hemispherical antennas must be used until the error can be corrected. This contingency must be addressed.

A series of link studies was performed to determine if acceptable communications can be established. We feel that for an operational program of this type, minimum link margins of +3 dB on downlink and +6 dB on uplink should be designed for. By assuming both operational constraints and improvements to the SCF remote tracking stations, all necessary links can be shown to be acceptable. The detailed calculations are in Appendix D.

The assumptions made for this conceptual study are detailed below. Other options are undoubtedly possible and should be addressed in the next phase of the program.

1. No ranging with low-gain spacecraft antennas. It appears that the ranging links cannot be supported by the hemispherical coverage antennas at maximum range. This constraint may complicate transfer trajectory operations since the high-gain antenna must be properly directed after some moderate range is exceeded.

2. No simultaneous command and ranging. The presence of the command tones robs the ranging signal of power in both the up and down links. The telemetry is also affected. Separation of command and ranging operations eliminates this degradation.

3. Command modulation index of 1.0 radian. The command modulation index can be either 0.3 or 1.0 radian. The higher value increases the command signal power by 9.4 dB.

4. Convolutional coding of telemetry. It is not practical to attempt low-power communications over this range without some form of forward error correction to strengthen the link. A rate-1/2, constraint length 7 convolutional encoding of the telemetry has been selected. For a ciphertext bit-error-rate of 10^{-6}, this reduces the required energy per bit/noise density from 10.6 dB to 5.1 dB, improving the link margin by 5.5 dB. The spacecraft hardware to encode is minimal. The decoders that will have to be added to the SCF remote stations are continually decreasing in cost.

5. Use of 10 kW transmitter. The use of the 10 kW transmitter rather than the 1 kW option increases the uplink power by 10 dB and is needed for uplink commanding to the low-gain antennas.

6. Improvement of the subcarrier demodulator. The loss associated with the subcarrier demodulator is given as 4.3 dB in the SCF manual (Ref. 7-1). More typical hardware would reduce this loss to 2.5 dB for a gain of 1.8 dB.

7. Reduction of carrier tracking bandwidth. Decreasing the carrier tracking bandwidth from 100 Hz to 30 Hz will increase the downlink carrier margin by 5.2 dB.

8. Improved 60-foot antenna performance. The SCF 60-foot antennas have a lower efficiency than the 46-foot antennas. If the larger reflectors could be upgraded to achieve the same efficiency, their use would improve the links by 2.3 dB.

9. Improved 60-foot receiving system. The 60-foot antennas have a higher system noise temperature than the 46-foot antennas. An upgrade of the 60-foot receivers to the same performance as the 46-foot system would improve the downlinks by 1.6 dB.

Assuming all the above assumptions are met (or their equivalent improvement in dB), the SIMPL worst case link margins at maximum range will be

	Command Uplink	Telemetry Downlink	Ranging
High-Gain S/C Antenna	24.1 dB	18.4 dB	2.3 dB
Low-Gain S/C Antenna	8.1 dB	2.7 dB	N/A

7.4 SPACECRAFT TRACKING AND ORBIT DETERMINATION

The orbit-tracking problem for SIMPL is similar to that for NASA's ISEE-3 spacecraft. NASA/Goddard's considerable experience with STDN (S-Band Tracking and Data Network) tracking support for ISEE-3 can help us determine requirements on the Air Force Satellite Control Facility (SCF). The typical STDN ground station for ISEE-3 used a 26 m antenna and a 125°K preamp. The lower G/T of the SCF stations relative to STDN is approximately compensated by the higher spacecraft antenna gain. Assuming the SCF tracking links are closed via the satellite's high-gain

antenna, it should be possible for the SCF S-Band SGLS tracking to realize similar performance. Characteristics of some of the best NASA tracking examples involved the following:

1. two week fit spans,

2. coherent ranging plus two-way doppler tracking,

3. two stations separated in latitude (one in the southern hemisphere and one in the northern hemisphere),

4. one 15-minute tracking contact per day per station, and

5. solution for the radiation pressure coefficient.

With these characteristics, NASA was able to hold 45-day predict errors below 10 km, one-sigma. Range biases at the 15 m one-sigma uncertainly level were the dominant error sources. Operational experience suggested fit spans could be shortened further, perhaps to one week. For that reason, we allowed adjustment of SIMPL's spin axis as frequently as every 11 days when we established the mission scenario.

The dependence on SCF's high gain ground antennas (46 or 60 ft) for tracking should not heavily penalize mission operations. It is expected that two or three 15-minute tracking passes per day would be more than ample. The total high-gain antenna usage should not exceed 45 minutes per day. These are tracking times; commanding and telemetry reception would be additional.

Based on the ISEE-3 experience, solution for the radiation pressure coefficient is recommended, with fit spans of 2-3 weeks. However, this coefficient might be determined with long arcs in the early phase of the mission, and then held fixed if much shorter fitting arcs (e.g., 7-11 day) are preferred between spin axis torquing operations.

SCF range biases at the 50 ft level are comparable to STDN range bias uncertainties for ISEE-3. The range noise level (20 ft one-sigma assumed here) is also comparable. The major difference appears to be in range rate. Whereas STDN two-way range rate is believed to be on the order of 0.1 cm/sec (0.0033 ft/sec), the SCF range rate noise level has been estimated at 0.15 ft/sec, almost 50 times higher. Generally, with a sufficiently high data rate, data random errors are not a problem, but this should be examined by covariance analyses in the next phase of the SIMPL program.

Based on NASA's experience with ISEE-3 and the above comparison between SCF and NASA tracking, we expect SCF could maintain ephemeris accuracy levels below 10-15 km between station keeping maneuvers. A preliminary covariance analysis was performed by the Aerospace Corporation at the request of JHU/APL, using SCF range and range rate tracking data from the New Hampshire and Indian Ocean stations. ISEE-3 orbit dimensions were used in this preliminary analysis. Orbit fit spans of both 7 and 14 day durations were examined, with 15 minutes per station per day. Measurements were obtained every 32 seconds during the tracking contacts. It was assumed that attitude control thruster forces would be balanced and not contribute any significant unmodeled translation forces. The radiation coefficient was solved for.

The analysis showed that interleaved tracking from two stations was necessary, and suggested that range data were far more important than range rate. When combined range and range rate data were taken from two stations, the position error in a 45-day prediction could be held to 8.0 km for a 14-day fit. Only half of that error was attributable to range noise. For a seven day fit, the position error increased to 11.6 km and was almost totally dominated by range noise.

In the next phase of the program, the foregoing tentative conclusions will be further tested in a series of covariance analyses. Exact SIMPL orbit dimensions will be used, and the effect of 7-11 day tracking arcs can be explored further. The various phases of the transfer orbit can be included also.

APPENDIXES

APPENDIX A: AMPTE SPACECRAFT DESCRIPTION

AMPTE stands for Active Magnetospheric Particle Tracer Explorers, a group of three spacecraft launched together to study interactions between the solar wind and the Earth's magnetosphere. The three spacecraft are the Ion Release Module (IRM), the Charge Composition Explorer (CCE), and a subsatellite. The three were launched by a single Delta 3924 in August 1984. The "AMPTE spacecraft" referred to in this report, and upon which SIMPL is based, is the AMPTE/CCE, designed and built by JHU/APL (see Figure A-1).

AMPTE/CCE was placed into a highly eccentric, low inclination orbit with an apogee of 8.8 Earth radii. It carries five instruments, including two designed and built by JHU/APL. These instruments measure magnetic field, plasma composition, plasma waves, and particle parameters. Measurements are made of both the unperturbed environment and in conjunction with ion releases performed by IRM.

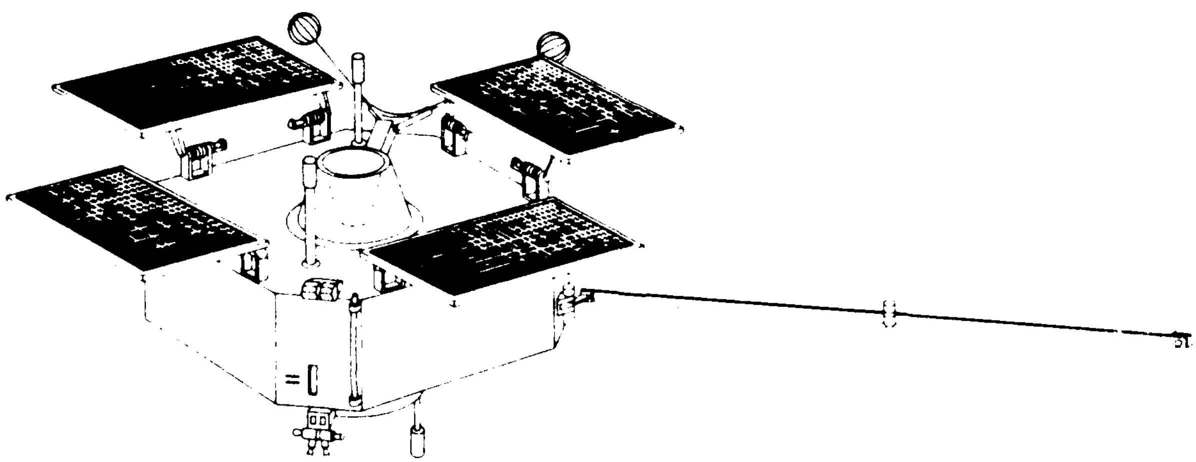


Fig. A-1 Charge Composition Explorer (CCE) of Active Magnetospheric Particle Tracer Explorers (AMPTE).

AMPTE was designed to be spin stabilized at 10 rpm and to keep the Sun within 10-30° of its Z-axis. This is accomplished by means of magnetic torquing based on attitude readings from a boom-mounted three-axis magnetometer and a DSAD. AMPTE also contained an integral solid rocket for inclination adjust. A cold-gas attitude control system was also included to provide rapid spin-up and attitude maneuvering early in the mission.

Power is provided by four deployed solar panels charging redundant 4-A-H, 28V Ni-Cd batteries and redundant main DC/DC converters. The array was designed to provide 140 W BOL and 100 W at the end of a four year mission. At launch the panels were folded against the sides of the spacecraft and held in place by the yo-yo cables and despin weights. Array output during injection was therefore limited.

The thermal design was based on use of multilayer blanket insulation, thermal coatings, active heaters, and four temperature-activated louvers mounted in critical locations. The thermal design was driven by the needs of the early orbit operations. Additional thermal protection was carried to protect against the solid rocket plume and isolate the rocket casing from the structure.

The structure consists of two honeycomb decks, upon which all the instruments and spacecraft electronics are mounted. These are supported by a load-bearing center column and further stiffened by radial struts and honeycomb side panels. The solid rocket motor and cold gas propellant tank are contained within the center column. AMPTE's launch weight was 242 kg.

Communication with AMPTE is accomplished through standard NASA S-band transponders with 2.5 W RF output. Four hemispherical coverage antennas are used, operating in switched pairs. A single tape recorder collects data at 3300 bps over long time spans and dumps it quickly (106 kbps) to NASA's Deep Space Network. Convolutional coding is employed on the downlinks. C&DH hardware is fully redundant in the RF and command portions, but only partially redundant in the telemetry portion.

The AMPTE orbit provides a severe radiation dose, so on-board electronics are designed to a general requirement of 100,000 rads. Electronics are also resistant to single event upsets caused by cosmic rays. Provisions were made to minimize electrostatic discharges.

AMPTE underwent a comprehensive test program including many of the special tests SIMPL will require, such as EMC, magnetics, and end-to-end compatibility testing with two different communications and control organizations. AMPTE/CCE has performed flawlessly since launch, meeting all its mission requirements.

APPENDIX B: SPACECRAFT RADIATION ENVIRONMENT

Knowledge of the natural radiation environment is necessary to determine sensor and solar cell degradation, select electronic parts, and establish shielding requirements. Our radiation environment analysis for SIMPL assumed a five-year mission at L_1 in addition to transfer orbit operations from the Shuttle. The five-year period was assumed to take place near Solar Maximum so contributions to the total radiation dose of solar protons from five anomalously large solar events (one per year) were included. This analysis is considered quite conservative. Subsequent to our analysis, the mission lifetime goal was extended to six years. The extra year would increase the figures presented here by less than 15%.

Table B-1 shows electron and solar proton integral energy spectra for the mission which can be used to predict solar panel damage. The NRL parameterization of anomalously large solar events was used for the solar proton fluences (Ref. B-1). The new AE 8 MAX model of Vette and colleagues at NASA/Goddard was used for transfer orbit electron fluences (Ref. B-2).

Tables B-2 and B-3 show the results of computing radiation doses at various depths for two different shielding geometries, using the SHIELDOSE computer code developed by Seltzer at NBS (Ref. B-3). The transmission surface of a finite aluminum slab is most accurate for shallow depths, while the center-of-aluminum-sphere model is most accurate for greater depths (>1 gm/cm^2) where protons coming from all directions dominate the total dose.

TABLE B-1

SIMPL particle radiation integral energy spectra

Duration: Five years at Earth-Sun libration point L_1

Epoch: Solar Maximum

Assumption: Five anomalously large solar events (one per year) plus electron flux in transfer orbit

ENERGY (> MEV)	SOLAR PROTONS (protons/cm^2)	TRAPPED ELECTRONS (electrons/cm^2)
0	1.22×10^{11}	
0.25		7.14×10^{12}
0.50		1.93×10^{12}
1	1.18×10^{11}	3.28×10^{11}
1.5		8.40×10^{10}
2	1.14×10^{11}	2.81×10^{10}
2.5		7.56×10^{9}
3		2.69×10^{9}
3.5		1.08×10^{9}
4	1.06×10^{11}	3.07×10^{8}
4.5		7.70×10^{7}
6	9.75×10^{10}	8.75×10^{5}
8	9.05×10^{10}	
10	8.40×10^{10}	
20	5.75×10^{10}	
40	2.71×10^{10}	
60	1.27×10^{10}	
80	6×10^{9}	
100	2.82×10^{9}	
150	4.26×10^{8}	
200	6.45×10^{7}	
250	9.80×10^{6}	
300	1.48×10^{6}	
350	2.25×10^{5}	
400	3.40×10^{4}	

Table B-2

SIMPL dose at transmission surface of finite aluminum slab shields

SHIELD DEPTH MILS	SHIELD DEPTH GRAMS/Cm^2	ELECTRON DOSE RADS (AL)	SOLAR PROTON DOSE RADS (AL)	TOTAL DOSE RADS (AL)
4.374	0.03	4.156×10^4	1.971×10^4	6.127×10^4
5.833	0.04	3.524×10^4	1.819×10^4	5.343×10^4
7.291	0.05	3.043×10^4	1.701×10^4	4.744×10^4
8.749	0.06	2.662×10^4	1.605×10^4	4.267×10^4
11.665	0.08	2.089×10^4	1.454×10^4	3.543×10^4
14.581	0.100	1.679×10^4	1.336×10^4	3.015×10^4
21.872	0.150	1.035×10^4	1.125×10^4	2.16×10^4
29.163	0.200	6.737×10^3	9.792×10^3	16529
43.744	0.300	3.125×10^3	7.839×10^3	10964
58.326	0.400	1.555×10^3	6.548×10^3	8103
72.907	0.500	8.082×10^2	5.616×10^3	6424
87.489	0.600	4.330×10^2	4.905×10^3	5338
116.652	0.800	1.326×10^2	3.875×10^3	4008
145.815	1.000	4.317×10^1	3.165×10^3	3208
182.269	1.250	1.117×10^1	2.535×10^3	2546
218.722	1.500	2.987	2.076×10^3	2079
255.176	1.750	< 1	1.736×10^3	1736
291.630	2.000	< 1	1.49×10^3	1469
364.537	2.500	< 1	1.090×10^3	1090
437.445	3.000	< 1	836	836
510.352	3.500	< 1	657	657
583.260	4.000	< 1	524	524
729.074	5.000	0	348	348

Table B-3

SIMPL dose at center of aluminum spheres

SHIELD DEPTH MILS	SHIELD DEPTH GRAMS/Cm^2	ELECTRON DOSE RADS (AL)	SOLAR PROTON DOSE RADS (AL)	TOTAL DOSE RADS (AL)
4.374	0.03	1.912×10^5	4.995×10^4	2.412×10^5
5.833	0.04	1.712×10^5	4.695×10^4	2.182×10^5
7.291	0.05	1.547×10^5	4.465×10^4	1.994×10^5
8.749	0.06	1.410×10^5	4.265×10^4	1.836×10^5
11.665	0.08	1.190×10^5	3.955×10^4	1.586×10^5
14.581	0.100	1.020×10^5	3.722×10^4	1.392×10^5
21.872	0.150	7.192×10^4	3.280×10^4	1.047×10^5
29.163	0.200	5.226×10^4	2.955×10^4	8.181×10^4
43.744	0.300	2.900×10^4	2.496×10^4	5.396×10^4
58.326	0.400	1.669×10^4	2.175×10^4	3.844×10^4
72.907	0.500	9.810×10^3	1.927×10^4	2.908×10^4
87.489	0.600	5.844×10^3	1.737×10^4	2.321×10^4
116.652	0.800	2.136×10^3	1.448×10^4	1.662×10^4
145.815	1.000	805	1.232×10^4	1.312×10^4
182.269	1.250	245	1.040×10^4	1.064×10^4
218.722	1.500	75	8.825×10^3	8900
255.176	1.750	23	7.675×10^3	7698
291.630	2.000	7	6.728×10^3	6735
364.537	2.500	< 1	5.215×10^3	5215
437.445	3.000	< 1	4.200×10^3	4200
510.352	3.500	0	3.460×10^3	3460
583.260	4.000	0	2.898×10^3	2898
729.074	5.000	0	1.998×10^3	1998

Figure B-1 shows the dose depth curves for the two geometry models. The dashed line represents an approximate true curve from the shallow depth, transmission surface of finite aluminum slab shield, to the greater depths at which the solar protons are dominant.

The analysis shows that at shallow depths, doses can be as great as 30,000-35,000 Rads (Al). From Table B-2 and Figure B-1:

DEPTH (mils)	DOSE (Rads (Al))
11.7	35,430
14.6	30,150

At greater depths the doses are reduced to 2000 to 13,000 Rads (Al). From Table B-3 and Figure B-1:

DEPTH (mils)	DOSE (Rads (Al))
146	13,120
729	2,000

Some benefit can be gained by increasing the shielding to 5 gm/cm^2 (approximately the maximum shield depth for light spacecraft) because the solar proton spectrum is softer than the trapped proton spectrum (average energy of the NRL parameterization is 26.5 MeV) and the fluences are not as large due to the few day limit on solar events.

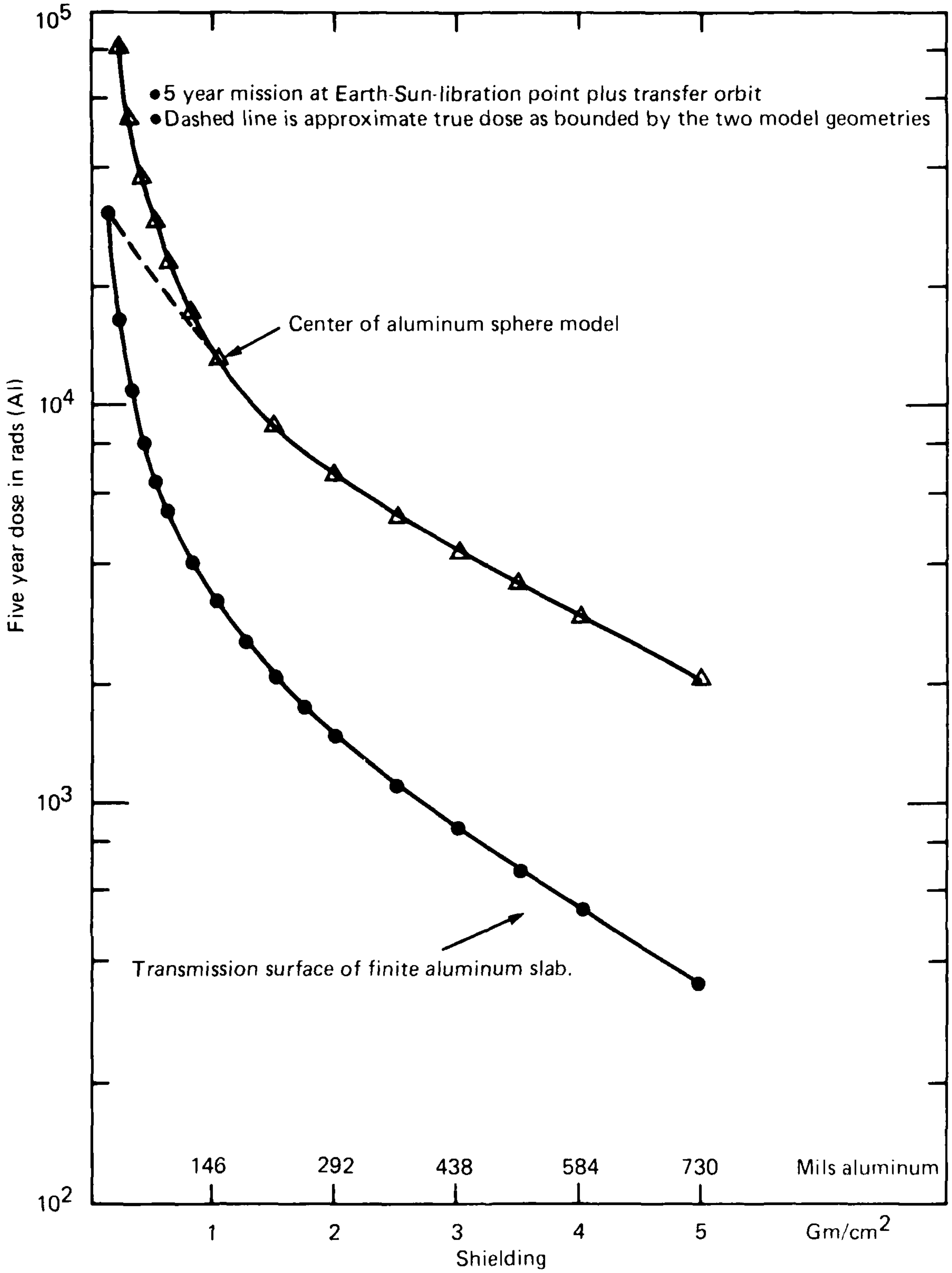


Fig. B-1. SIMPL dose-depth curve.

We conclude that for engineering design purposes, the worst case dose for SIMPL (for an exposed instrument, for example) would be about 35,000 Rads (Al). The irreducible dose would be about 2000 Rads (Al), near the center of the spacecraft.

APPENDIX C: DETAILED WEIGHT AND POWER SUMMARY

The following two pages show detailed weight and power budgets for SIMPL. Power is shown as both peak and orbit average based on the average subsystem duty cycles shown. The letter key next to the peak power indicates whether the source is the unregulated (battery) bus (U), the regulated 28 V bus (R), or the main converter (M). Different losses are carried for each case. A power growth allowance of 20% is carried.

Weights shown are individual JHU/APL estimates based on heritage from similar programs and subcontractor inputs for purchased items. Harness and balance weights are estimated based on prior JHU/APL experience. Total structure is estimated at 15% of total subsystem weights, confirmed independently by examining AMPTE weights and estimating the conical adapter section. A weight growth allowance of 20% would be desirable at this early stage of the program, and is carried on all items except the actual liquid hydrazine (since the tanks are already filled to capacity). Also shown are the excess PAM-D loft capability, and the total delta-v realized from the on-board propulsion system design for the wet weight shown.

"SIMPL"

AFGL "SIMPL" SPACECRAFT SUMMARY......OCT 16,1985

(......This is SIMPL-on-AMPTE w/ 4 GOES tanks....)

S/C Subsystem	Power(W)	× Duty Cycle	= Av. Power(W)	Weight(kg)	Comments
Attitude Control					
Redun. Active Nutation Damper	2.0 M	0.100	0.2	1.0	no spin ripple damper
(2) Passive Nutation Dampers				2.0	= 4 x AMPTE weight
(2) Accelerometers	0.2 M	0.100	0.0	0.2	
Redundant Sun Sensor	1.2 M	0.100	0.1	incl.	
Star Scanner #1	1.9 R	0.100	0.2	3.0	one powered
Star Scanner #2	1.9 R	0.000	0.0	3.0	
Redun. Att. Preprocessor	2.0 M	1.000	2.0	1.3	
Redun. Sector Clk/Thrust Ctrl/ & Sun Sensor Elex	1.0 M	1.000	1.0	1.5	
Propulsion (4 GOES-type tanks)					
(4) Tanks, empty wt.				7.5	
(12) Thrusters				6.0	Ham.-Std. type thrusters
Plumbing, valves, misc.	36.0	0.000	0.0	3.3	valve power TBD
Liquid Hydrazine				123.0	maximum N2H4 capacity
Pressurant				1.7	
Power (DET system + solar-only backup)					
Solar Array (225 BOL/191 EOL W.)				10.9	AMPTE size + 16%
Battery (1 x 6AH, 18 cells)	4.0	1.000	4.0	6.5	incl. mtg. plate
Redund. BCR & Control Units	1.0	1.000	1.0	2.4	incl. shunt drivers
Battery Charger	0.8	1.000	0.8	1.5	
Boost Regulator	0.8	1.000	0.8	3.0	
Redund. Main Converters	6.0 R	1.000	6.0	2.0	effic. = 85%
Overcur/Undervolt, htrs, misc	0.3	1.000	0.3	0.2	
RF Communications					
Antennas (4 hemi-omnis)				2.3	
Antenna (1 small dish)				3.5	
(2) Pre-mod processors,one on	0.5 M	1.000	0.5	incl.	AMPTE design
(2) Xpndrs, Rcv. Mode	9.0 R	1.000	9.0	3.8	
(2) 5 W. Xmtrs, one on	27.8 R	1.000	27.8	3.5	
(1) diplexer	0.0	0.000	0.0	0.5	
RF Switching	5.0 R	0.000	0.0	1.1	
RF coaxial cables	0.0	0.000	0.0	1.5	.250" low-loss cable
Cmd. & Data Handling					
Dual Cmd. Processor & PSM	6.4 U	1.000	6.4	12.5	GEOSAT proc wt, new PSM
Dual Cmd. Converter				1.8	GEOSAT x .7
(2) Command Auth. Units	1.4	1.000	1.4	0.6	both powered
Dual Telem. Processor	4.0 M	1.000	4.0	7.6	GEOSAT processor wt. + TCG
(2) Telem. Encryptors	0.7 M	1.000	0.7	0.6	one powered
Dual Oscillator	1.5 M	1.000	1.5	1.1	
Time Management Unit	1.0	1.000	1.0	incl.	
Shuttle Interface	5.0 M	0.000	0.0	1.5	
Thermal					
Subsystem MLI Blankets				3.0	
Subsys. Heaters & Misc.	10.0 R	1.000	10.0	0.5	
Boom Sensors Thermal Control	1.0 R	1.000	1.0	0.1	
Single Battery Louver				0.8	

	W.		W.	kg	
Instruments					each with dc/dc converter
Outboard Magnetometer Sensor	0.1 R	1.000	0.1	0.3	
Inboard Magnetometer Sensor	0.1 R	1.000	0.1	0.3	is also backup unit
Redund. Magn. Electronics	2.0 R	1.000	2.0	4.0	
Magnetometer Boom (20 ft.)				7.3	MAGSAT boom less gimbal
Magn. Boom Harness				0.5	
Inertia Boom				2.5	tape boom + endmass??
Solar Wind Sensor Head	0.5 R	1.000	0.5	2.0	
Solar Wind Electronics #1	3.5 R	1.000	3.5	3.0	one on
Solar Wind Electronics #2	0.0 R	1.000	0.0	3.0	
Particle Detector	5.0 R	1.000	5.0	5.0	
Radio Wave Instrument (body)	5.0 R	1.000	5.0	3.0	
(2) Radio Wave Preamps	0.6 R	1.000	0.6	0.4	
(2) Radio Wave Wire Antennas	4.2	0.000	0.0	5.0	Fairchild ISEE-type
On-board Processor #1	5.0 M	1.000	5.0	4.0	
On-board Processor #2	5.0 M	0.000	0.0	4.0	
Harness & Flight Plugs				22.0	low-EMI (AMPTE + 30%)
Balance Weights				10.0	
SUBSYSTEM SUBTOTALS	163.4		101.5	302.6	
Structure @ 15%				45.4	AMPTE pri+sec structure = 30 kg conical adapter = 12 kg????
Growth Allowance (20%)*	32.7		20.3	45.0	
"SIMPL" SPACECRAFT TOTALS	196.1		121.8	393.0	
PAM-D Lift Capability to Orbit***				608.0	
PAM-D Lift Margin***				215.0	

Orbit Average Power, Watts

Wet Weight, kg

R = Reg. Bus (28 V. +/-2%)   U = Unreg. Critical Bus (20-29 V.)   M = Main Converter

115.6 % growth balances lift capability*

NOTE: growth not carried on liquid hydrazine

Delta-v for 393.0 kg (wet weight) spacecraft is 791.5 m/s**

***off-loaded (baseline) PAM-D

OCT 16,1985 / EJHOF

APPENDIX D: RF LINK ANALYSES

The following six tables show the detailed analyses for the major RF links discussed in Section 7.3.

Table D-1

SIMPL UPLINK WITH HIGH GAIN S/C ANTENNA 10/30/85

MODE: COMMAND
ORBIT TYPE: HALO ABOUT LIBRATION POINT
GROUND ANTENNA: 18.28 meter ELEVATION: 5 Deg.
RANGE: Max. 1.7 Mmeter
FREQUENCY: Carrier: 1.803 Ghz; Cmd Tones(kHz) 95 76 65
Sync: 500 Hz Triangular Wave
Ranging Clock: 500 kHz. PRN RATE: 1 Mchip
COMMAND BIT RATE: 1 kbps
MODULATION: PCM
MODULATION INDEX: Command 1 Radian. Ranging 0 Radian
CONSTANTS: Jo(CMI) 0.7652 Cos^2 1
J1(CMI) 0.4400 Sin^2 0

PARAMETER	CONDITION	VALUE	UNITS
GROUND TRANSMITTER POWER	10 kWATTS	70.0	dBm
GROUND ANTENNA GAIN (RHCP)	60 Feet Dia.	42.7	dBic
POTENTIAL ANTENNA GAIN IMPROVEMENT:		4.6	dB
EFFECTIVE RADIATED PWR.		112.7	dBm
PATH LOSS		-222.2	dB
ATMOSPHERIC LOSS		-0.3	dB
POLARIZATION LOSS		-0.5	dB
POINTING LOSS		0.0	dB
SATELLITE ANTENNA GAIN	(Parabolic with RHCP)	14.0	dBic
SATELLITE PASSIVE LOSS		-1.5	dB
MAX. TOTAL RECEIVED PWR.		-93.2	dBm
SAT. ANTENNA NULL DEPTH		0.0	dB
MIN. TOTAL RECEIVED PWR.		-93.2	dBm
RECEIVER NOISE FIGURE		5.0	dB
SYSTEM NOISE TEMPERATURE		1003.2	deg-K
SYSTEM NOISE DENSITY		-168.6	dBm/Hz
CARRIER/TOTAL POWER		-2.3	dB
RECEIVED CARRIER POWER		-95.5	dBm
CARRIER NOISE BANDWIDTH	2000 Hz	33.0	dB-Hz
NOISE POWER		-135.6	dBm
CARRIER/NOISE POWER		40.1	dB
REQUIRED CARRIER/NOISE PWR		12.0	dB
CARRIER MARGIN		28.1	dB
COMMAND PWR/TOTAL PWR		-4.1	dB
RECEIVED COMMAND PWR.		-97.3	dBm
PREDETECTION NOISE BW	2 kHz	33.0	dB-Hz
PREDETECTION NOISE PWR.		-135.6	dBm
PREDETECTION SNR		38.3	dB
COMMAND DATA RATE	1000 bps	30.0	dB-bps
AVAIL. ENERGY/BIT/NOISE DENSITY		41.3	dB
DECODER DEGRADATION		3.0	dB
REQ'D ENERGY/BIT/NOISE DENSITY		14.2	dB

BER=1*10E-6
AVAILABLE COMMAND MARGIN 24.1 dB

Table D-2

SIMPL UPLINK WITH HIGH GAIN S/C ANTENNA 10/30/85

MODE: RANGING
ORBIT TYPE: HALO ABOUT LIBRATION POINT
GROUND ANTENNA: 18.28 meter ELEVATION: 5 Deg.
RANGE: Max. 1.7 Mmeter
FREQUENCY: Carrier: 1.803 Ghz; Cmd Tones(kHz) 95 76 65
Sync: 500 Hz Triangular Wave
Ranging Clock: 500 kHz. PRN RATE: 1 Mchip
COMMAND BIT RATE: 1 kbps
MODULATION: PCM
MODULATION INDEX: Command 0 Radian. Ranging 0.3 Radian
CONSTANTS: Jo(CMI) 1 Cos^2 0.9127
J1(CMI) 0 Sin^2 0.0873

PARAMETER	CONDITION	VALUE	UNITS
GROUND TRANSMITTER POWER	10 kWATTS	70.0	dBm
GROUND ANTENNA GAIN (RHCP)	60 Feet Dia.	42.7	dBic
POTENTIAL ANTENNA GAIN IMPROVEMENT:		4.6	dB
EFFECTIVE RADIATED PWR.		112 7	dBm
PATH LOSS		-222.2	dB
ATMOSPHERIC LOSS		-0.3	dB
POLARIZATION LOSS		-0.5	dB
POINTING LOSS		0.0	dB
SATELLITE ANTENNA GAIN	(Parabolic with RHCP)	14.0	dBic
SATELLITE PASSIVE LOSS		-1.5	dB
MAX. TOTAL RECEIVED PWR.		-93 2	dBm
SAT. ANTENNA NULL DEPTH		0.0	dB
MIN. TOTAL RECEIVED PWR.		-93.2	dBm
RECEIVER NOISE FIGURE		5.0	dB
SYSTEM NOISE TEMPERATURE		1003.2	deg-K
SYSTEM NOISE DENSITY		-168.6	dBm/Hz
CARRIER/TOTAL POWER		-0.4	dB
RECEIVED CARRIER POWER		-93.6	dBm
CARRIER NOISE BANDWIDTH	2000 Hz	33 0	dB-Hz
NOISE POWER		-135.6	dBm
CARRIER/NOISE POWER		42.0	dB
REQUIRED CARRIER/NOISE PWR		12 0	dB
CARRIER MARGIN		30.0	dB
RANGING PWR/TOTAL PWR		-10.6	dB
RECEIVED RANGING PWR		-103.8	dBm
RANGING CHANNEL BANDWIDTH	900 kHz	59 5	dB-Hz
NOISE PWR IN RANGE CH.		-109.0	dBm
RECV'D RANGE PWR/NOISE PWR.		5.3	dB

Table D-3

SIMPL UPLINK COMMAND WITH LOW GAIN S/C ANTENNA

MODE: COMMAND (Without Ranging)
ORBIT TYPE: HALO ABOUT LIBRATION POINT
GROUND ANTENNA: 18.28 meter ELEVATION: 5 Deg.
RANGE: Max. 1.7 Mmeter
FREQUENCY: Carrier: 1.803 GHz; Cmd Tones(kHz) 95 76 65
Sync: 500 Hz Triangular Wave
Ranging Clock: 500 kHz. PRN RATE: 1 Mchip
COMMAND BIT RATE: 1 kbps
MODULATION: PCM
MODULATION INDEX: Command 1 Radian. Ranging 0 Radian
CONSTANTS: Jo(CMI) 0.7652 Cos^2 1
J1(CMI) 0.4400 Sin^2 0

PARAMETER	CONDITION	VALUE	UNITS
GROUND TRANSMITTER POWER	10 kWATTS	70.0	dBm
GROUND ANTENNA GAIN (RHCP)	60 Feet Dia.	42.7	dBic
POTENTIAL ANTENNA GAIN IMPROVEMENT:		4.6	dB
EFFECTIVE RADIATED PWR.		112.7	dBm
PATH LOSS		-222.2	dB
ATMOSPHERIC LOSS		-0.3	dB
POLARIZATION LOSS		-0.5	dB
POINTING LOSS		0.0	dB
SATELLITE ANTENNA GAIN (Quadrifiliar Helix with RHCP)		-2.0	dBic
SATELLITE PASSIVE LOSS		-1.5	dB
MAX. TOTAL RECEIVED PWR.		-109.2	dBm
SAT. ANTENNA NULL DEPTH		0.0	dB
MIN. TOTAL RECEIVED PWR.		-109.2	dBm
RECEIVER NOISE FIGURE		5.0	dB
SYSTEM NOISE TEMPERATURE		1005.4	deg-K
SYSTEM NOISE DENSITY		-168.6	dBm/Hz
CARRIER/TOTAL POWER		-2.3	dB
RECEIVED CARRIER POWER		-111.5	dBm
CARRIER NOISE BANDWIDTH	2000 Hz	33.0	dB-Hz
NOISE POWER		-135.6	dBm
CARRIER/NOISE POWER		24.1	dB
REQUIRED CARRIER/NOISE PWR		12.0	dB
CARRIER MARGIN		12.1	dB
COMMAND PWR/TOTAL PWR		-4.1	dB
RECEIVED COMMAND PWR.		-113.3	dBm
PREDETECTION NOISE BW	2 kHz	33.0	dB-Hz
PREDETECTION NOISE PWR.		-135.6	dBm
PREDETECTION SNR		22.3	dB
COMMAND DATA RATE	1000 bps	30.0	dB-bps
AVAIL. ENERGY/BIT/NOISE DENSITY		25.3	dB
DECODER DEGRADATION		3.0	dB
ENERGY/BIT/NOISE DENSITY		14.2	dB

BER=1*10E-6
AVAILABLE COMMAND MARGIN 8.1 dB

Table D-4

SIMPL DOWNLINK TO SCF WITH HIGH GAIN S/C ANTENNA

```
MODE:              TELEMETRY & RANGING
ORBIT TYPE:         HALO AT LIBRATION POINT
GROUND ANTENNA:          DIA.     18.5 Meters;   ELEVATION:       5 deg
                         GAIN:     48.2 dB
RANGE:                   MAX.  1700000 km;
FREQUENCY:          CARRIER:     2.252 GHz;          SUBC. 1  1.024 MHz
BIT RATE:                                            SUBC. 1    0.1 kbps
UPLINK RNG'ING/NOISE PWR          5.3 dB;            RATIO:    3.39
MODULATION. (PCM+PCM/PSK)/PM
MOD  INDEX.         RANGING        0.3 Radians       SUBC. 1    1.4 Radian
                    COMMAND          0 Radians
TURNAROUND RATIO:               1.1
EFFECT. RNG'ING. MOD  IND       0.264 Radians
CONSTANTS:             Jo(Xr)   0.983                  Jo(Xs)  0.567
                       J1(Xr)   0.131                  J1(Xs)  0.542
                    Jo(Xc)^2        0 dB
EFFECT. NOISE MOD. INDEX        0.143 Radians;
```

PARAMETER	CONDITION	VALUE	UNITS
S/C TRANSMITTER POWER:	5 Watts	37.0	dBm
S/C PASSIVE LOSS:		-1.5	dB
S/C ANTENNA GAIN:	Parabolic (RHCP)	14.0	dBic
EIRP:		49.5	dBm
REMOD. NOISE LOSS:		-0.1	dB
DISPERSIVE LOSS:		-224.1	dB
ATMOSPHERIC LOSS:		-0.3	dB
POLARIZATION LOSS:		-0.5	dB
GND RECEIVE ANTENNA GAIN:	18.5 Meters, (60')	48.2	dBic
POTENTIAL ANTENNA GAIN IMPROVEMENT:		1.6	dB
RECEIVER PASSIVE LOSS:		0.0	dB
POINTING LOSS:		-0.1	dB
TOTAL RECEIVED POWER:		-125.8	dBm
GND SYS G/T (SGLS)		24.8	dB
ANTENNA NOISE TEMPERATURE:	(Sun in sidelobes)	50.0	deg-K
SYSTEM NOISE TEMPERATURE:	(Requires upgrade - to 46')	220.0	deg-K
SYSTEM NOISE DENSITY:		-174.3	dBm/Hz
CARRIER PWR/TOTAL PWR:		-5.1	dB
CARRIER POWER REC'D:		-130.9	dBm
CARRIER TRACKING BW:	30 Hz	14.8	dB-Hz
CARRIER/NOISE POWER REC'D		28.6	dB
CARRIER/NOISE PWR REQ'D:		6.0	dB
CARRIER MARGIN:		22.6	dB
RANGE SIG. PWR/TOTAL PWR:		-19.6	dB
RANGING PWR REC'D:		-145.4	dBm
RANGE DET. BW:	1 Hz	0.0	dB-Hz
RANGING PWR/NOISE PWR:		28.9	dB
RNGE PWR/NOISE PWR REQ'D:		26.6	dB
RANGING MARGIN:		2.3	dB
SUBC1 DATA PWR/TOTAL PWR:		-2.5	dB
SUBC1 DATA PWR REC'D:		-128.3	dBm
SUBC1 DATA RATE:	0.1 kbps	20.0	dB-Hz
SUBC1 DATA PWR/NOISE PWR:		26.0	dB
SUBC1 DEMOD LOSS:		2.5	dB
SUBC1 DATA/NOISE PWR REQD	BER- 10E-6	5.1	dB
Assumes Rate 1/2, length 7, Convolution Decoder			
DATA MARGIN:		18.4	dB

Table D-5

SIMPL DOWNLINK TO SCF WITH LOW GAIN S/C ANTENNA

MODE: TELEMETRY (During absence of uplink commands & ranging)
ORBIT TYPE: HALO AT LIBRATION POINT
GROUND ANTENNA: DIA. 18.5 Meters; ELEVATION: 5 deg
GAIN: 48.2 dB
RANGE: MAX. 1700000 km;
FREQUENCY: CARRIER: 2.252 GHz; SUBC. 1 1.024 MHz
BIT RATE: SUBC. 1 0.1 kbps
MODULATION: (PCM+PCM/PSK)/PM
MOD. INDEX: RANGING: 0 Radians SUBC. 1 1.4 Radian

CONSTANTS: Jo(x) 0.567
J1(x) 0.542

PARAMETER	CONDITION	VALUE	UNITS
S/C TRANSMITTER POWER:	5 Watts	37.0	dBm
S/C PASSIVE LOSS:		-1.5	dB
S/C ANTENNA GAIN:	Quadrifiliar Helix (RHCP)	-2.0	dBic
EIRP:		33.5	dBm
DISPERSIVE LOSS:		-224.1	dB
ATMOSPHERIC LOSS:		-0.3	dB
POLARIZATION LOSS:		-0.5	dB
GND RECEIVE ANTENNA GAIN:	18.5 Meters, (60')	48.2	dBic
POTENTIAL ANTENNA GAIN IMPROVEMENT:		1.6	dB
RECEIVER PASSIVE LOSS:		0.0	dB
POINTING LOSS:		-0.1	dB
TOTAL RECEIVED POWER:		-141.7	dBm
GND SYS G/T (.GLS)		24.8	dB
ANTENNA NOISE TEMPERATURE:	(Sun in sidelobes)	50.0	deg-K
SYSTEM NOISE TEMPERATURE:	(Requires upgrade - to 46')	220.0	deg-K
SYSTEM NOISE DENSITY:		-174.3	dBm/Hz
CARRIER PWR/TOTAL PWR:		-4.9	dB
CARRIER POWER REC'D:		-146.7	dBm
CARRIER TRACKING BW:	30 Hz	14.8	dB-Hz
CARRIER/NOISE POWER REC'D		12.9	dB
CARRIER/NOISE PWR REQ'D:		6.0	dB
CARRIER MARGIN:		6.9	dB
SUBC1 DATA PWR/TOTAL PWR:		-2.3	dB
SUBC1 DATA PWR REC'D:		-144.0	dBm
SUBC1 DATA RATE:	0.1 kbps	20.0	dB-Hz
SUBC1 DATA PWR/NOISE PWR:		10.3	dB
SUBC1 DEMOD LOSS:		2.5	dB
SUBC1 DATA/NOISE PWR REQD	BER= 10E-6	5.1	dB
Assumes Rate 1/2, length 7, Convolution Decoder			
DATA MARGIN:		2.7	dB

Table D-6

SIMPL DOWNLINK TO AWS
(In presence of ranging)

```
MODE:            TELEMETRY (only)
ORBIT TYPE:       HALO AT LIBRATION POINT
GROUND ANTENNA:       DIA.         5 Meters;   ELEVATION:      5 deg
                     GAIN:        39 dB
RANGE:                MAX.   1700000 km;
FREQUENCY:        CARRIER:     2 252 GHz;        SUBC. 1   1.024 MHz
BIT RATE:                                        SUBC. 1     0.1 kbps
UPLINK RNG'ING/NOISE PWR:        5.3 dB;         RATIO:    3.39
MODULATION:     PCM/PSK1/PM
MOD. INDEX:       RANGING:       0.3 Radians     SUBC. 1     1.4 Radian
                  COMMAND:         0 Radian
EFFECT. RNG'ING. MOD. IND      0.264 Radians
CONSTANTS:            Jo(X)    0.983                  Jo(x)  0.567
                      J1(x)    0.131                  J1(x)  0.542
EFFECT. NOISE MOD. INDEX:      0.143 Radians;
```

PARAMETER	CONDITION		VALUE	UNITS
S/C TRANSMITTER POWER:	5 Watts		37.0	dBm
S/C PASSIVE LOSS:			-1.5	dB
S/C ANTENNA GAIN:	DISH	(RHCP)	14.0	dB
EIRP:			49.5	dBm
REMOD. NOISE LOSS:			-0.1	dB
DISPERSIVE LOSS:			-224.1	dB
ATMOSPHERIC LOSS:			-0.3	dB
POLARIZATION LOSS:			-0.5	dB
GND RECEIVE ANTENNA GAIN:	5 Meters	(16.5')	39.0	dBic
RECEIVER PASSIVE LOSS:			0.0	dB
POINTING LOSS:			-0.1	dB
TOTAL RECEIVED POWER			-136.6	dBm
ANTENNA NOISE TEMPERATURE		(SUN IN SIDELOBES)	135.0	deg-K
RECEIVER NOISE TEMP			91.0	deg-K
SYSTEM NOISE TEMPERATURE			226.0	deg-K
SYSTEM NOISE DENSITY			-175.1	dBm/Hz
GND SYS G/T			15.5	
CARRIER PWR/TOTAL PWR			-5.1	dB
CARRIER TRACKING BW	30 Hz		14.8	dB-Hz
CARRIER/NOISE POWER REC'D			18.6	dB
CARRIER/NOISE PWR REQ'D			10.0	dB
CARRIER MARGIN			8.6	dB
SUBC1 DATA PWR/TOTAL PWR.			-2.5	dB
SUBC1 DATA RATE	0.1 kbps		20.0	dB-Hz
SUBC1 DATA PWR/NOISE PWR			16.0	dB
SUBC1 DEMOD LOSS:			2.5	dB
SUBC1 DATA/NOISE PWR REQD	BER= 10E-6		5.1	dB
RATE 1/2 CONVOLUTIONAL ENCODING:				
DATA MARGIN			8.4	dB

APPENDIX E: SUN NOISE EFFECTS ON SOON/RSTN ANTENNAS

The SIMPL spacecraft is to orbit the L_1 libration point and continuously transmit data. The ground station antennas will be pointing close to the Sun, so Sun generated noise may enter through a sidelobe and increase the noise temperature of the receiving system. A calculation has been made which indicates that the additional noise temperature should be less than 70°K with a typical ground station for the orbit dimensions selected.

The contribution to antenna noise temperature from an extended source is found by integrating the product of the antenna gain and the source temperature over the sphere about the antenna (Ref. E-1).

$$T_a = \frac{1}{4\pi} \int_{\phi=0}^{2\pi} \int_{\theta=0}^{\pi} G(\theta,\phi)\, T(\theta,\phi) \sin\theta \, d\theta d\phi$$

where T_a = added noise temperature

$G(\theta,\phi)$ = gain of the antenna in the θ,ϕ direction

$T(\theta,\phi)$ = temperature of the source in the θ,ϕ direction

θ,ϕ = standard spherical coordinates

At S-band the black body temperature, T, can be related to its brightness, B, by the Rayleigh-Jeans approximation to Planck's law:

$$T = \frac{\lambda^2}{2K} B$$

where B = brightness, watts/(m^2 Hz rad^2)

T = temperature, °K

K = Boltzman's constant, 1.38×10^{-23} Joule/°K

λ = free space wavelength, m

The noise temperature addition in terms of brightness is therefore:

$$T = \frac{\lambda^2}{8\pi K} \int_{\phi=0}^{2\pi} \int_{\theta=0}^{\pi} G(\theta,\phi)\, B\,(\theta,\phi) \sin\theta \, d\theta d\phi$$

The Sun subtends an angle of about 0.5°. The surface integral can be limited to this portion of the sphere. Since both antenna gain and brightness are non-negative quantities, an upper bound on the noise temperature can be found by using the maximum antenna gain which is directed at the Sun.

$$T \leq \frac{\lambda^2}{8\pi K} G_S \int_{\phi=0}^{2\pi} \int_{\theta=0}^{\pi} B(\theta,\phi) \sin\theta \, d\theta d\phi$$

where G_S = maximum antenna gain directed at the Sun.

The integral is now simply that of the brightness, which is the total flux density of the Sun. The units generally employed are solar flux units [1 SFU = 10^{-22} watts/(m^2Hz)].

$$T_a \leq \frac{\lambda^2}{8\pi K} G_S\, S(10^{-22})$$

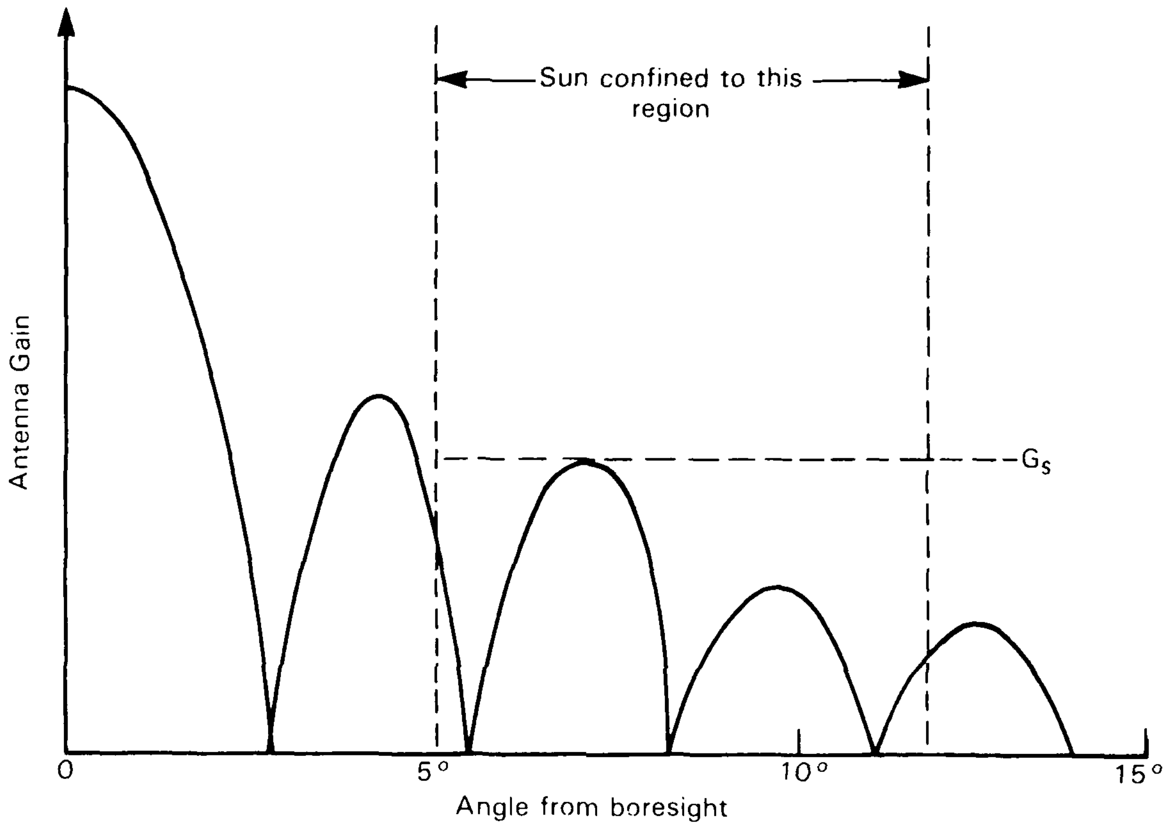
where S = flux density in SFU.

The center of the telemetry band is 2250 MHz, which corresponds to a free space wavelength of 0.133 meter. The maximum flux density is estimated at 1,000 SFU, based on worst-case solar maximum conditions. Substituting these quantities yields,

$$T_a \leq 5.13\ G_S$$

The angle between boresight of the receiving antenna and the edge of the Sun varies from approximately 5.0° to 11.6° with the proposed SIMPL orbit (Fig. E-1). An acceptably low antenna gain can be maintained in this critical region without a special design. For example, JHU/APL has a 5-meter backup antenna at its Satellite Tracking Facility. No difficult sidelobe specifications were placed on this autotracking antenna. An examination of its principal plane radiation patterns indicates a maximum gain of 11.3 dBi (G_S = 13.49) in the critical region. An upper bound on the Sun noise for such an antenna is therefore $T_a \leq 69.2^oK$. A Sun temperature of 70° has been included in the link calculations. A receiving antenna designed for lower sidelobes could substantially reduce this source of noise.

Alternate missions can be conceived which are not limited to an S-band downlink. An X-band link would suffer even less from Sun noise. For a given size receiving antenna, the boresight gain would be increased but the critical region would be farther out on the sidelobe structure, generally yielding a lower absolute gain. In addition, the noise temperature of the Sun is lower at this higher frequency.



G_S is the maximum antenna gain in the critical region.

Fig. E-1. Effect of noise from the Sun on reception at SIMPL ground station.

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6-1 NASA NHB 1700.7A, "Safety Policy and Requirements for Payloads Using the Space Transportation System."

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GLOSSARY

ACS	Attitude Control System
AFGL	Air Force Geophysics Laboratory
AFGWC	Air Force Global Weather Central
A-H	Ampere-Hour
AMPTE	Active Magnetospheric Particle Tracer Explorers
ASE	Airborne Support Equipment
AWS	Air Weather Service
BOL	Beginning of Life
bps	bits per second
CADH	Command and Data Handling
CCE	Charge Composition Explorer
CMOS	Complementary Metal-Oxide-Semiconductor
COMSEC	Communications Security
CT	Ciphertext
DET	Direct Energy Transfer
DoD	Department of Defense
DOD	Depth of Discharge
DSAD	Digital Solar Attitude Detector
DSN	Deep Space Network (NASA)
EMC	Electromagnetic Compatibility
EMI	Electromagnetic Interference
EOL	End of Life
ESMC	Eastern Space and Missile Center
FSK	Frequency Shift Keying
FSS	Fine Sun Sensor
GN_2	Gaseous Nitrogen
GSE	Geocentric Solar Ecliptic, Ground Support Equipment
GSFC	Goddard Space Flight Center

GWC	Global Weather Central
IIL	Current Injection Logic
ISEE	International Sun-Earth Explorer
I_{sp}	Specific Impulse
ISTP	International Solar Terrestrial Physics Program
JHU/APL	Johns Hopkins University/Applied Physics Laboratory
JSC	Johnson Space Center
KSC	Kennedy Space Center
LNA	Low Noise Amplifier
MCC	Mission Control Complex, Mission Control Center
Mcps	Million chips per second
MLI	Multi-layer Insulation
N_2H_4	Monopropellant Hydrazine
NMOS	N-channel Metal-Oxide-Semiconductor
NSA	National Security Agency
OUSS	Overcurrent and Undervoltage Sensing Switch
PAM	Payload Assist Module
PAS	Panoramic Attitude Sensor
PRN	Pseudorandom Noise
PROM	Programmable Read Only Memory
PSM	Power Switching Module
RHCP	Right Hand Circularly Polarized
RSTN	Radio Solar Telescope Network
SCF	Satellite Control Facility
SDSAD	Spinning Digital Solar Attitude Detector
SEU	Single Event Upset
SFU	Solar Flux Units
SGLS	Space Ground Link Subsystem

SIMPL	Synoptic Interplanetary Monitor Platform at L_1
SOON	Solar Observing Optical Network
SPDT	Single Pole Double Throw
sps	Symbols per second
SSSS	Solid State Star Scanner
STDN	Space Tracking and Data Network (NASA)
STS	Space Transportation System (Shuttle)
TBD	To be determined
TDRSS	Tracking and Data Relay Satellite System
TMU	Time Management Unit
WAMPTE	WIND-on-AMPTE
WIND	the L_1 orbiting spacecraft of ISTP

END